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Helicopter Fatigue

A Review of Current Requirements and Substantiation Procedures

NORTH ATLANTIC TREATY ORGANIZATION

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Five papers presented at the 47th Meeting of the Structures and Materials Panel held in Florence, Italy on 25–29 September 1978.

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PREFACE

Helicopter designers have always been concerned with fatigue phenomena and the experience developed in this field in NATO countries has resulted in fatigue requirements which appear to have very similar objectives and philosophies.

The five papers collected in this Report were presented in Florence to the Working Group on Helicopter Fatigue of the Structures and Materials Panel, during the 47th Meeting of the Panel.

This Report may be considered as a detailed and valuable review of current fatigue requirements and substantiation procedures in the United States, United Kingdom, Germany, Italy and France in the field of Helicopter Fatigue.

Fatigue specialists are aware of the rather uncomfortable situation which is reflected through these papers; although general requirements and specifications seem to be very similar, approved procedures applied by manufacturers may sometimes appear to be rather arbitrary or, in some cases, to differ significantly from one firm to another.

The material collected in this publication must be considered as a helpful survey to be used by helicopter specialists with a view to intensifying cooperative action within the NATO community towards improvement, rationalization and standardization of helicopter service life prediction.

J.M.FEHRENBACH
Chairman, Working Group on
Helicopter Fatigue

CONTENTS

U.S. ARMY HELICOPTER FATIGUE REQUIREMENTS AND SUBSTANTIATION PROCEDURES

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SUMMARY

This paper provides the current fatigue criteria and testing requirements for U.S. Army helicopter structures with primary emphasis on dynamic components. Related to these requirements are the various applications applied by the individual helicopter contractors in meeting these requirements. The comparative industry applications of the requirements were brought to light as a result of the Army's latest major helicopter competitions for the Utility Tactical Transport Aircraft System (UTTAS), recently designated "BLACK HAWK", and the Advanced Attack Helicopter (AAH). These competitions resulted in evaluations by the Army of four major helicopter companies (two competitors for each program) and provided significant "lessons learned" in future Army fatigue requirements primarily because of differences in loads application, S/N curves shape criteria, working level curve deviations and component testing techniques applied by each contractor. The paper will address in detail these differences and how they relate to the current Army requirements. As a result of these "lessons learned", the Army is in the process of specifying new fatigue requirements for future helicopter procurements. This paper will identify those requirement changes that have been initiated and outline those changes that are currently under study. Careful attention is needed in attempting to "tighten up" a requirement since a delicate trade off exists between fatigue substantiation based on individual contractor historical methods and capability versus required standardization of these methods. One cannot ignore each contractors' experience and success in using his own developed methods.

RECENT ARMY REQUIREMENTS

In the interest of time, this section will discuss the Army helicopter fatigue requirements for the BLACK HAWK and the AAH only since they represent the recent Army philosophy for modern battlefield helicopters. The discussion will be centered around the general methods required, specific life requirements and the mission spectra specified.

For dynamic components, the Army requires the load or stress versus cycles to failure (S/N) method of testing and verification. A minimum of six test specimens is required to establish the mean curve with the reduction factors left to the discretion of the individual contractor based on his individual experience but approved by the Army prior to the design stage of the components. The standard Miners Rule Method is required for life determination based on the Army specified mission spectrum as will be discussed later in the section. A detailed discussion of the S/N curve shapes and reduction factors used by the various contractors will also be discussed later including specific reduction factors specified by the Army for interim component life. The necessity for interim component lives was due to the length of the competitive development program in which detailed Army flight evaluations were conducted during the "fly offs" of these competitions. Due to either lack of time, monetary restraints or possibility of redesign, the full six specimens were not required for interim life determination. Therefore, conservative S/N mean reduction factors were specified for those cases in which one or two specimens were tested.

Due to the complexity of the designs and the cost involved, spectrum type testing was specified for major airframe components.

Life Requirements: As can be seen from Table 1, the minimum required fatigue life for all dynamic components is 5000 hours for the UTTAS BLACK HAWK and 4500 hours for the AAH, references 1 and 2. For comparative purposes, original requirements for the existing corresponding class of U.S. Army helicopters are also shown, references 3 and 4. From a pure requirements standpoint, one can easily see the more demanding requirements for the Army's next generation helicopters. The airframe, while no life is specified, does require no major overhaul in less than the hours shown in Table 1 for the new systems and once again provides a significant improvement over previous requirements.

Since this is a paper on requirements, the actual life achievement of the various aircraft components is not presented, but it is important to note that in the new designs the actual lives significantly exceed the requirements shown in Table 1. This is primarily due to the stringent fail safe survivability/invulnerability requirements which in many cases design the components with a high fatigue life as a fallout. It should also be noted that in many components the actual lives of the UH-1 and AH-1 also exceed the requirements originally stated.

Mission Spectrum: Presented in Figures 1 and 2 are the airspeed and load factor distribution portions of the Army's utility helicopter mission spectrum. The figures show a comparison between the most recently developed utility helicopter, the BLACK HAWK, and its predecessor, the UH-1. It should be noted that the airspeed distribution has been normalized in terms of V_H to provide a direct comparison as a fraction of each aircraft's maximum speed capability. As shown in the table in Figure 1, the V_H of the BLACK HAWK is substantially higher than that of the UH-1 which, independent of the distribution of airspeeds creates a more severe fatigue requirement. Figures 3 and 4 present similar comparison data for the two generations of attack helicopter, the AAH and the AH-1.

Several important differences in the spectra of the two generations of aircraft should be noted. The first is that the UH-1 and the AH-1 were developed without an Army defined mission spectrum. The spectra of the BLACK HAWK and the AAH, however, were based on actual helicopter usage data obtained in Southeast Asia (References 5 and 6). Combining the "lessons learned" data from Southeast Asia with the expected mission profiles defined by the Army for their next generation helicopter systems resulted in the airspeed and load factor distribution shown in Figures 1 thru 4. These distributions were then augmented with several other important mission spectrum parameters as presented in Table 2. A second major difference can be seen in Figures 1 and 3. The BLACK HAWK and the AAH have a substantially greater percentage of time allocated to the high speed region. Additionally, the AAH has an increased time allocation in the low speed regime reflecting its nap-of-the-earth mission. A final difference is seen in Figures 2 and 4 where increased load factor requirements denote the need for greater maneuver capability.

U.S. HELICOPTER INDUSTRY COMPARATIVE APPLICATION

It can be seen from a review of the last section on specific Army requirements that in many instances each contractor is left with applying his individual methods of fatigue analyses and testing techniques in arriving at final lives. This section will discuss the more important areas in which the applications differ between contractors based on their individual experience.

Load Application: The ideal method of measuring flight loads would result in valid statistical loads data. Normally this implies the need for long-term sampling. To obtain good data it is essential that the instrumentation is reliable and data collection and extraction is simple and not ambiguous. However, this luxury cannot be attained during a development program. Cost and time do not allow for long-term measurement of flight loads. Fatigue life calculations for helicopter dynamic components are based on the one time measurement of loads obtained for the flight conditions specified in the Flight Loads Survey. This approach obviously results in subjective utilization and interpretation of the flight loads data.

Initial usage of these measured data is usually very conservative. However, as the calculated fatigue lives become critical with respect to the design requirements, the conservatism is whittled away sometimes at almost an alarming rate. But, to begin at the beginning, even before the Flight Loads Survey commences, compromises are required on the measured loads data. All the strain gages required to accurately and precisely measure the load environment for each dynamic component cannot be installed. Restrictions are imposed because of structural geometric barriers and also because of practical limitations of the recording and data handling and reduction instrumentation systems. The next step is to compress the fatigue mission spectrum to be covered by the Flight Loads Survey. Program time and money constraints do not allow for measurement of loads at all the possible flight conditions and configuration variations. It is attempted to define and include the most critical conditions and most probable configurations with spot checks of other areas as an additional safeguard. Normally, the intent is to collapse the fatigue mission spectrum for the Flight Loads Survey in a conservative manner but the net result is a decrease in accuracy in the measurement of "mission loads".

Another significant limitation is the fact that only one Flight Loads Survey is conducted; consequently, statistical loads variations are not contained in the data. Factors contributing to the loads scatter, which are not accounted for are aircraft differences, pilot techniques, and weather conditions. Also, the assumption is made during design that gusts and high load factors will not occur simultaneously. The Flight Loads Survey is then conducted under calm weather conditions to preclude encountering significant gusts while pulling high load factors. Only one American prime helicopter manufacturer accounts for the scatter in loads when calculating the fatigue lives of dynamic components. The method is a complex statistical procedure employing Monte Carlo techniques using ten thousand random data points and a confidence level of 99.95%.

After the Flight Loads Survey has been completed, the manner of applying the measured loads is subject to many variations. The first almost universally applied assumption is that the maximum peak load acts over the duration of the maneuver. However, as straight forward as this application seems, it is sometimes applied differently. Figure 5 represents the loads data measured during a maneuver. The peak oscillating load is usually interpreted as $\pm 1/2\ x_p$. This is the oscillating load that is applied for the duration of the maneuver for the percentage of time that this maneuver occurs in the fatigue mission spectrum. Some companies are even more conservative and use $\pm 1/2\ x_p'$ or $\pm 1/2\ x_p$ (see Figure 5) whichever is highest.

When the calculated fatigue lives do not meet expectations, the very conservative approach explained in the previous paragraph is abandoned and all companies resort to cycle counting. This technique of loads application can have a profound impact on the calculated

fatigue lives sometimes increasing them by factors of three or four. Cycle counting can be accomplished in various ways which are more or less conservative. Figure 6 illustrates the three more commonly applied cycle counting techniques. The block technique (Figure 6A) is the most conservative because it assumes the peak load under each envelope block occurs for the duration of the block. Actual cycles are not counted but the time of each block is used as the percentage of occurrence. The interval technique (Figure 6B) counts the peaks between intervals and assumes that they occur at the highest level of the intervals. This is very similar to the block technique but less conservative since the intervals used are more refined and actual cycles are counted. Figure 6C is the exact peak counting technique and consequently is the least conservative. It counts all peaks above the endurance limit and uses these loads throughout the fatigue mission spectrum for the percentage of time the particular maneuver is encountered.

The Goodman Correction is applied to account for the impact of steady loads on the fatigue lives of dynamic components. Normally a mean (steady) load is applied during the bench fatigue test to minimize or preclude the correction required to be made to measured flight loads data. The mean load can vary significantly for the various flight conditions which is not accounted for in the bench test. Also, the mean load applied during bench testing is usually based on analysis. There is no hard and fast rule as to when a Goodman Correction must be applied to the measured flight loads data. This is another subjective area which can have a significant influence on the calculated fatigue lives but is primarily left to the discretion and engineering judgment of the fatigue analyst.

S/N Curve Shapes: A major consideration in the fatigue life evaluation of a component is the development of its S/N curve. This curve is formulated by testing to failure six of the components and then applying a characteristic curve shape to these data. The difficulty arises in the fact that the industry has not agreed upon a standard set of characteristic curves. For example, Table 3 presents for three common metals the total number of curve shapes and their respective endurance limits used by various contractors in recent Army development programs. It can be seen that the number of curve shapes for a particular metal can range from 1 to 6 while the endurance limits may vary as much as one order of magnitude.

The development of S/N curves for composite materials involves several additional complicating factors. The first of these is the fact that the shape of a composite material S/N curve is very flat compared to a metallic curve and is, therefore, very sensitive to the value of oscillating load. This sensitivity requires that considerable precision be used when establishing an S/N curve for a composite material. A second complication involves the number of composites which can be produced. In general, composite material properties are a function of the fiber, matrix, processing and surface finish. Since each of these determinants has many variations, it is possible to produce a very large number of composite substances. The development of S/N curves for this many materials would be an exceedingly difficult task. A third problem is the difficulty in defining the failure point of a composite material. In general, composites do not fail catastrophically; they typically exhibit a loss of stiffness when exposed to oscillating loads above the endurance level. They may even delaminate without any significant loss in strength. The problem, therefore, is in determining an acceptable level of stiffness degradation which in turn will probably depend on how the component is utilized in the aircraft.

Working Level Curve Derivation: Another major area in which a significant difference exists between contractor application is the reduction factors used in reducing the mean S/N curve to a working level curve for establishment of final endurance limits. The significance of this variation is shown in Table 4 for four contractors involved in recent major Army helicopter development programs. The reduction factors are expressed in terms of Mean (M) minus a given statistical variation (3σ) of the data or a standard percentage. In those cases in which more than one reduction factor is shown for a given material, the largest factor is used in the final determination. These factors are also derived by the individual contractor based on his experience.

As discussed in the introduction, the Army requires establishment of interim fatigue strength for each component for new or modified designs during their development flight testing. Before first flight of a new helicopter the Army requires a minimum of one specimen of each component to be bench tested with a conservative reduction factor specified for fatigue calculation. Table 5 presents the reduction factors required for components with either one or two specimens tested. In those cases where more than two specimens have been tested, the reduction factor reverts to the contractors' individual application as shown in Table 4.

Component Testing Techniques: The laboratory bench fatigue test techniques employed to obtain S/N data for dynamic components can induce significant variations in test results. Differences in data points will impact the mean and working level S/N curves and consequently the resulting calculated fatigue lives. Although the American Helicopter Industry generally uses similar test procedures and philosophies, it is believed that the subtle differences are significant enough to warrant a brief discussion in the paragraphs which follow.

Care must be taken in the test setup to insure that the loads are properly induced and that the reaction of the test specimen is thoroughly monitored. If the control of the test is fully automated, the applied loads can be held to close tolerances and cracks can be detected in early stages. When control is primarily achieved by manual techniques

and crack detection is accomplished primarily by visual inspection, the accuracy of test results is usually reduced. The number of cycles required for a crack to propagate from initiation to complete fracture can be in the tens of thousands, consequently, the earlier the crack is detected the more conservative is the resulting S/N curve.

The magnitude, phasing and degree of simulation of the actual loads can have a profound influence on test results. Usually the initial oscillatory loads are established as a percentage of predicted level flight loads such as 200% at cruise speed. If run out is attained at the initial load level, the magnitude of the loads is increased and testing is continued. This process is repeated until a failure is detected in the test specimen. Contractors vary in the application of this procedure and in the utilization of the resulting data. Some increase the loading in large increments while others use small increments. Also, some companies take credit for the cycles accumulated at the lower load levels when plotting the S/N curves and others ignore them.

Phasing and degree of simulation of the applied loads have a strong influence on the failure modes. Many companies take the conservative approach and assume initially that all loads are in phase. As Flight Loads Survey data becomes available the phasing of the test loads are usually adjusted. Some companies apply calculated phasing relationships and adjust these based on measured data. This approach involves a certain degree of risk because the calculations can be unconservative. If a load is predicted to be small and not included in the original test setup it is usually costly and time consuming to modify the test fixture. This situation is encountered very frequently particularly on rotor blades because rotor load prediction programs are in dire need of improvement. Many companies attempt to verify analytically that the measured loads will not significantly impact the established failure modes.

Two areas which are in need of immediate attention are the establishment of failure criteria for composite materials and a realistic method of applying Ground-Air-Ground (GAG) cycles during rotor component bench fatigue testing. Failure criteria for metals is almost universally accepted as crack initiation. However, since composites are very forgiving and have many redundant load paths and by construction natural crack stoppers, the definition of failure becomes very difficult to formulate. Some engineers believe it should be based on a detectable change in stiffness while others think it should be based on the inability to continue to carry load. Very conservative members of the technical community believe it should be based on initial cracks, debonding or interlaminar shear detection.

There are many methods in existence in the industry to account for GAG load applications. One is to test one specimen applying S/N loads and a second specimen applying GAG cycle loads only. The test results are then compared to determine which failure modes are most critical and how the remaining specimens should be tested. Another is to apply GAG cycles to a specimen that has reached run-out during S/N testing to demonstrate adequate fatigue strength under both types of loading. Still another method involves testing a specimen with GAG cycle loading until failure or a large number of cycles are accumulated. The required GAG cycles to attain a design fatigue life is divided by the accumulated number of test cycles to obtain a damage fraction due to GAG cycle loading. The GAG cycle loading damage fraction is then added to the S/N damage fraction in calculating the final fatigue life of the dynamic component.

IMPROVEMENTS IN FUTURE ARMY FATIGUE REQUIREMENTS

As can be seen from the discussions in the paper, the Army has made significant improvements in their fatigue requirements for new helicopter designs, specifically in the minimum life and mission spectrum areas. However, based on "lessons learned" in recent competitive development programs, it has become very obvious that additional definitive requirements will be necessary for future Army developments. The lack of these improvements in the current designs in no way jeopardizes their safety and/or capability. Each contractors' application and methods have been carefully reviewed and approved by the Army. The disadvantage in this approach is the engineering manhours required to evaluate each contractors' methods and their relationship to the specified requirements. In many cases this results in over-conservatism and added program costs which could be reduced with the standardization of methods imposed by the Army for future developmental programs. Future fatigue requirements are summarized in two categories; (1) short range - those that can be implemented immediately in the next procurement with little impact on the contractors' established methods and (2) long range - those that require detailed study and evaluation prior to implementation since they could have a significant impact on contractor methods, as follows:

Category 1: Short Range

Mission spectrum improvements including definitive:

- Gust application
- Marriage of airspeed and load factor percent of occurrence
- GAG cycle definition and allocation
- Improved weapons firing spectra
- Maneuver description

Flight loads and component testing improvements including definitive:

- Cycle counting techniques
- Bench test loads application
- Bench test loads phasing
- Goodman corrections

Category 2: Long Range

- Flight loads variation allocation
- Composite materials failure criteria
- S/N curve shape standardization
- S/N reduction factor standardization
- GAG cycle bench test standardization

CONCLUSIONS

The method used to establish the fatigue lives of dynamic components by each American helicopter company is primarily based on their particular experiences and design philosophies. It has been derived through an iterative process and even today is undergoing change. Each company has been successful in applying their methodology to the development of dynamic components. In the past, success has been measured strictly by safety; for example, if no premature failures occurred due to design deficiencies, the method is successful.

With the advent of sophisticated mission requirements, which are pushing the state-of-the-art in helicopter performance and payload capability, it is time to re-examine the needs and objectives in determining fatigue lives. Safety cannot be compromised because failure of dynamic components is almost always catastrophic. On the other hand, excessive costs and weight penalties which are the direct result of excessive conservatism can no longer be tolerated.

It is believed that the materials, equipment, tools and know how is available in the aerospace industry today to develop a standardized method of establishing fatigue lives of dynamic components. The objectives of the standardized method as a minimum must be to eliminate excessive conservatism thereby reducing the cost and weight of dynamic components without compromising safety.

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TABLE 1
MINIMUM FATIGUE LIFE REQUIREMENTS
MAJOR ARMY PROGRAMS

HELICOPTER SYSTEM	DYNAMIC COMPONENTS	AIRFRAME
UTTAS (BLACKHAWK)	5000 HRS	NO OVERHAUL IN LESS THAN 5000 HRS
UH-1	1000 HRS	NOT SPECIFIED
AAH	4500 HRS	NO OVERHAUL IN LESS THAN 4500 HRS
AH-1	1000 HRS	NOT SPECIFIED

TABLE 2
MISSION SPECTRUM PARAMETERS

- MANEUVER SYMMETRY DISTRIBUTION
  - 60% OF MANEUVERS NOT SYMMETRICAL
  - 40% OF MANEUVERS SYMMETRICAL
- CLIMB AND DESCENT DISTRIBUTION FOR LEVEL FLIGHT/MODERATE MANEUVERING
  - 5% IN ASCENDING MODE
  - 8% IN DESCENDING MODE
- GROSS WEIGHT LIFE DISTRIBUTION
  - 40% AT BASIC STRUCTURAL DESIGN GROSS WEIGHT (BSDGW)
  - 40% AT BSDGW MINUS PAYLOAD AND 50% FUEL
  - 20% AT MAXIMUM ALTERNATE GROSS WEIGHT
- MANEUVER LIFE DISTRIBUTION
  - 70% LEVEL FLIGHT WITH MODERATE MANEUVERING
  - 20% PULL-UP AND TURN MANEUVERS
  - 10% CONTROL REVERSAL, AUTOROTATION, AND ACCELERATION/DECELERATION
- MISSION ORDNANCE (MO) DISTRIBUTION
  - 810 ANTIARMOR MISSIONS WITH 55% OF MO EXPENDED IN HOVER
  - 875 ESCORT MISSIONS WITH 20% OF MO EXPENDED IN HOVER
  - REMAINING MO DISTRIBUTED PER FORWARD FLIGHT SPECTRUM

TABLE 3
METAL S/N CURVE SHAPE
COMPARISON

CON	TOTAL NUMBER OF CURVE SHAPES			ENDURANCE LIMIT (CYCLES)		
	AL	STL	TI	AL	STL	TI
A	1	3	1	10^8	10^7	10^7
B	6	3	6	5×10^7	10^7	5×10^7
C	1	1	1	5×10^7	10^7	10^7
D	2	2	1	10^8	10^8	10^8

TABLE 4
FINAL MEAN S/N REDUCTION FACTORS

CONTRACTOR	ALUMINUM	STEEL	TITANIUM
A	M– 3σ OR M–25%	M– 3σ OR M–20%	
B	M – 3σ, M–20% OR BOTTOM OF SCATTER		
C	M – 3σ OR M–18%		
D	M–39%	M–30%	M–30%

TABLE 5
INTERIM MEAN S/N
REDUCTION FACTORS

MATERIAL	1 SPECIMEN	2 SPECIMENS
ALUMINUM	M–50%	M–35%
STEEL	M–50%	M–30%
TITANIUM	M–50%	M–30%

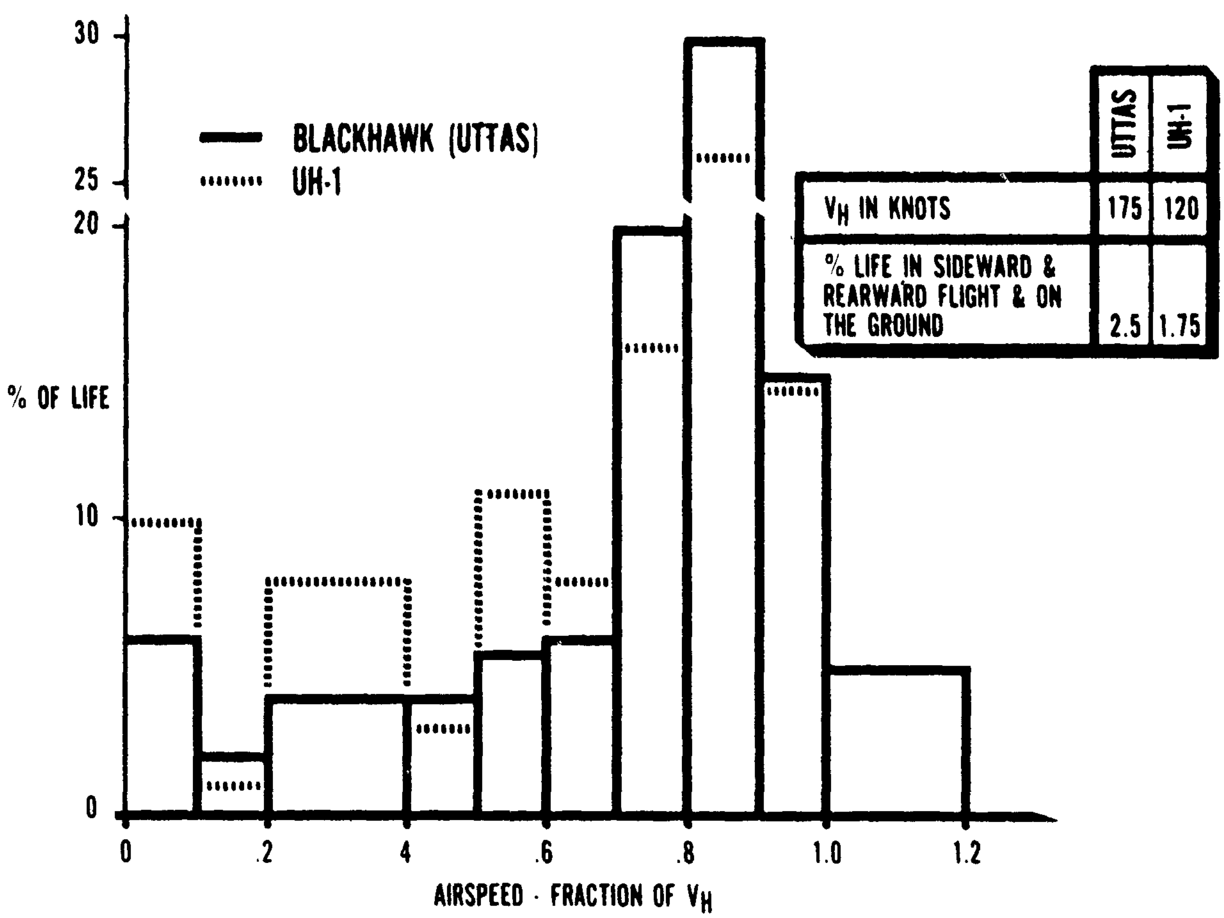


FIGURE 1
UTILITY HELICOPTER MISSION SPECTRUM
AIRSPEED DISTRIBUTION

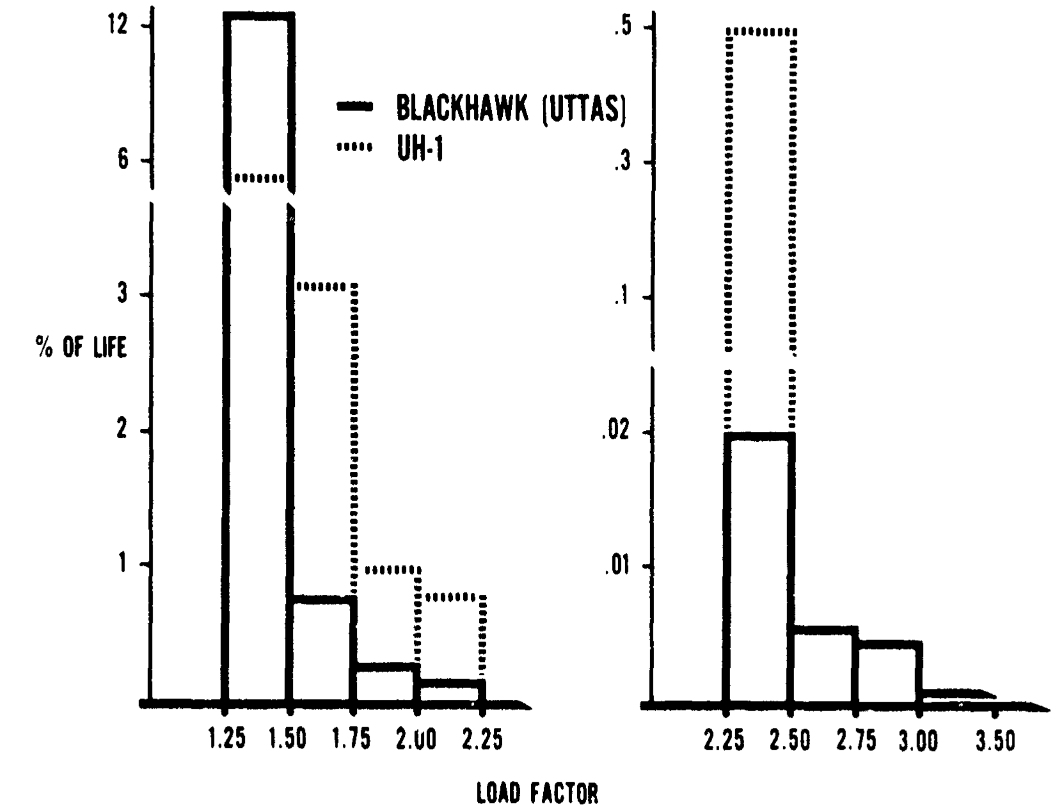
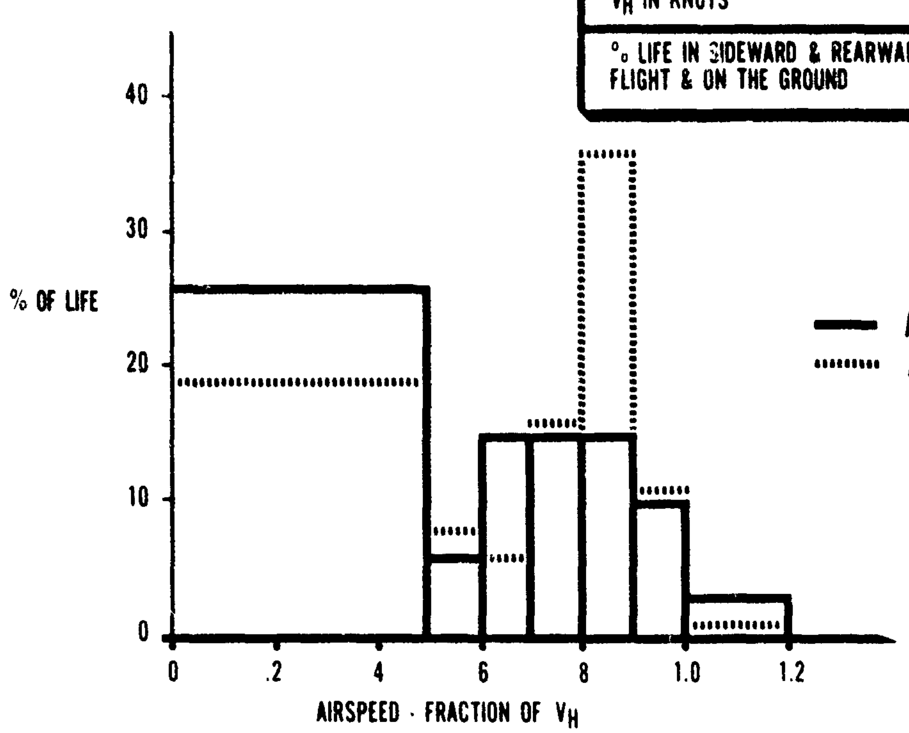


FIGURE 2
UTILITY HELICOPTER MISSION SPECTRUM
LOAD FACTOR DISTRIBUTION



	AAH	AH-1
V_H IN KNOTS	164	130
% LIFE IN SIDEWARD & REARWARD FLIGHT & ON THE GROUND	10	3

FIGURE 3
ATTACK HELICOPTER MISSION SPECTRUM
AIRSPEED DISTRIBUTION

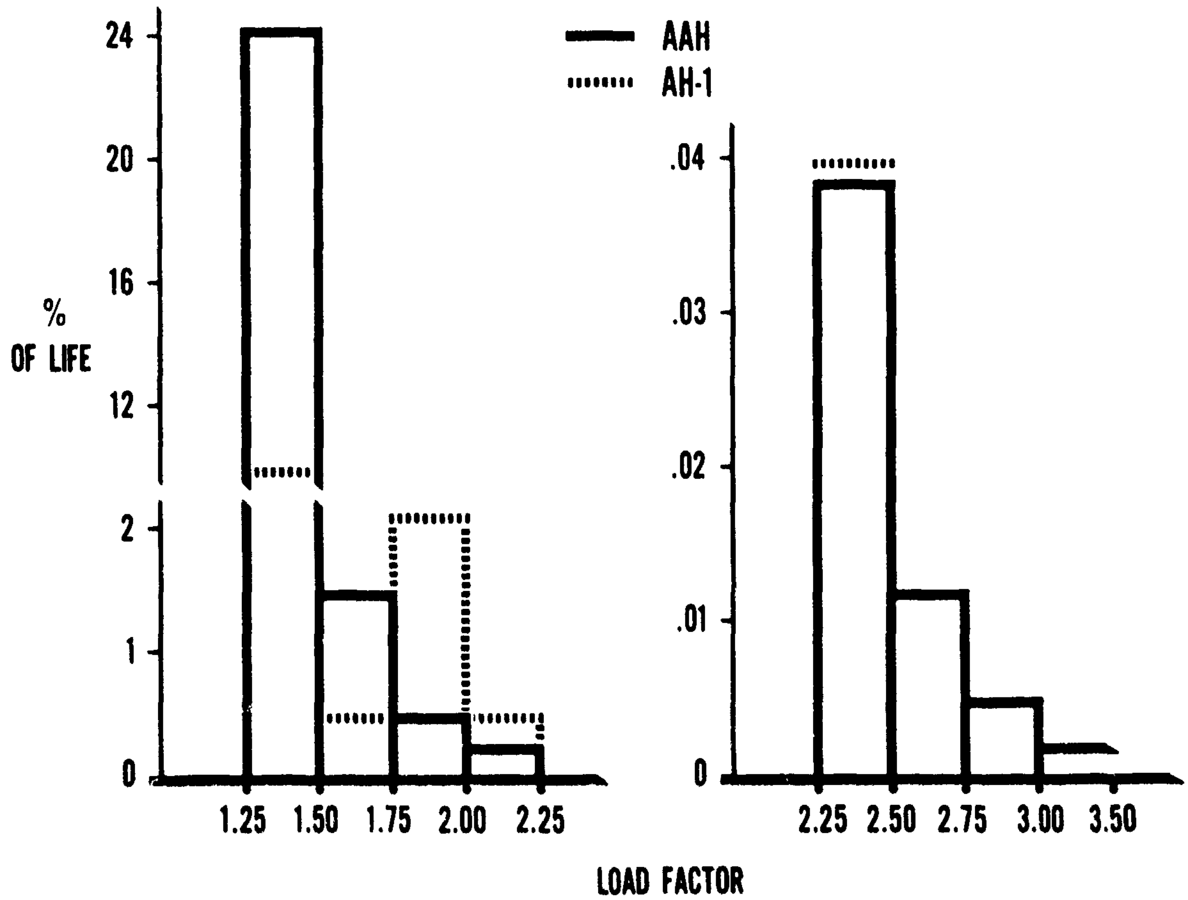


FIGURE 4
ATTACK HELICOPTER MISSION SPECTRUM
LOAD FACTOR DISTRIBUTION

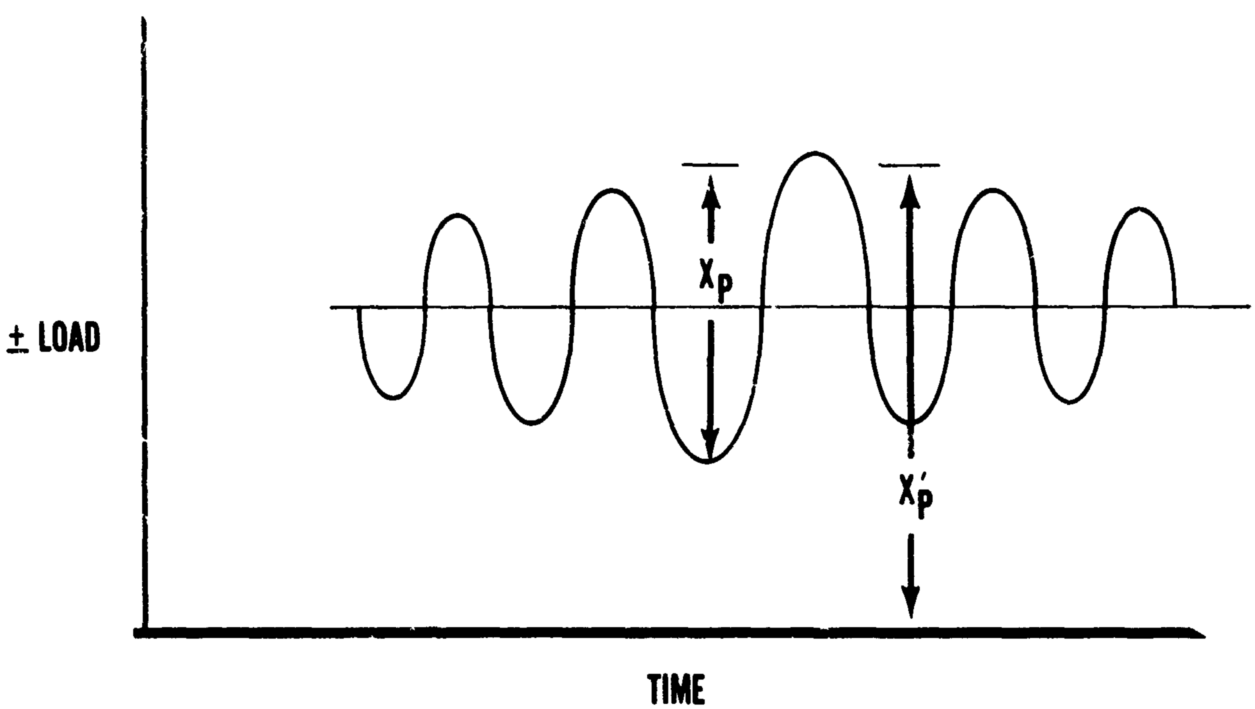


FIGURE 5
TYPICAL DYNAMIC COMPONENT FLIGHT LOAD TIME HISTORY
LOAD MAGNITUDE

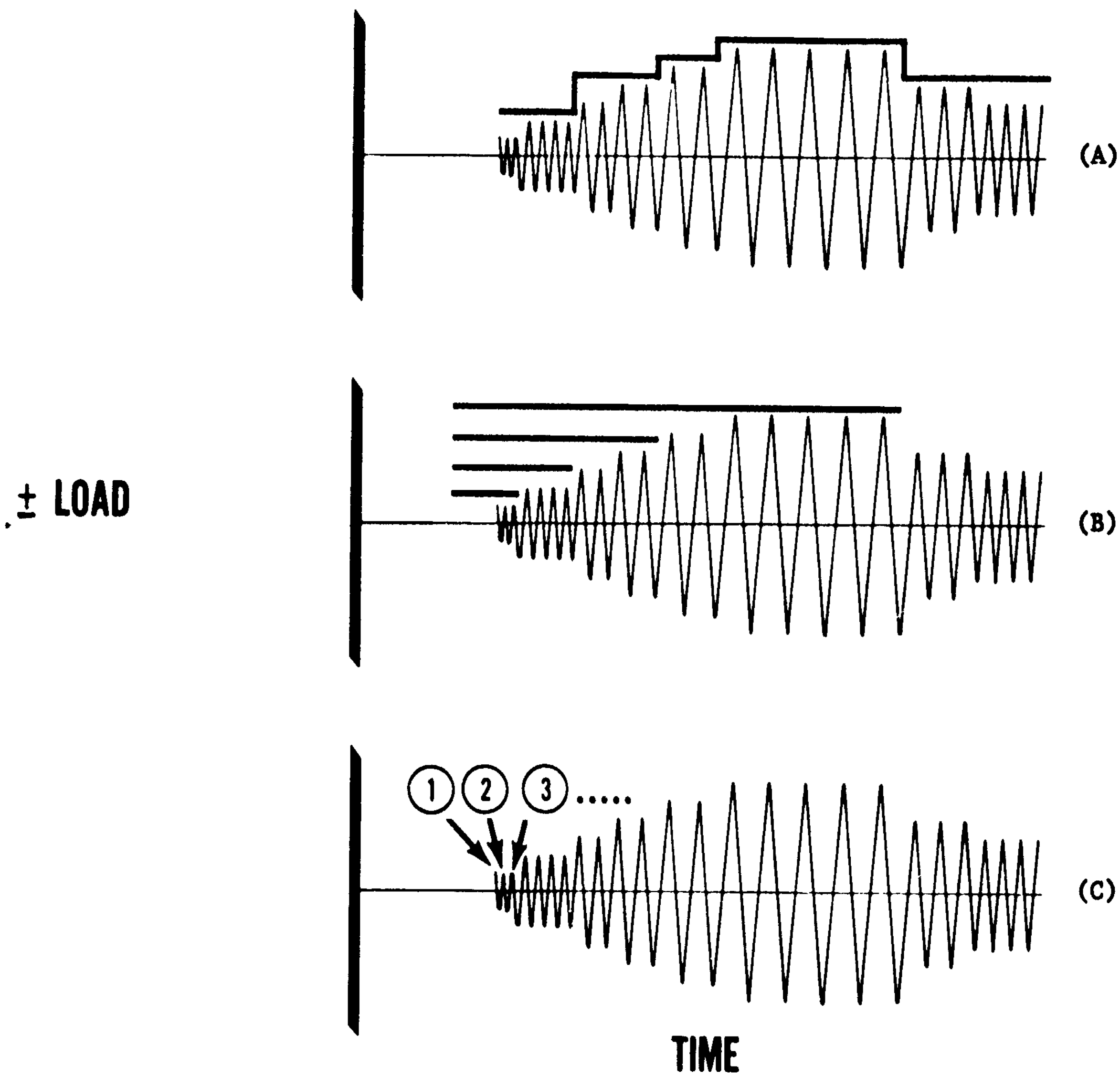


FIGURE 6
TYPICAL DYNAMIC COMPONENT
FLIGHT LOAD TIME HISTORY
(CYCLE COUNTING)

HELICOPTER FATIGUE EVALUATION
(The U.K. Approach)

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Summary

This paper outlines the approach to helicopter fatigue evaluation currently adopted in the U.K. The philosophies of fatigue substantiation have been used satisfactorily by Westland Helicopters Limited for the Lynx and it is considered that the practicability of the approach has been well established.

The main part of the paper is concerned with the safe fatigue life substantiation of the "Vital" components of a helicopter and consideration is given to three phases in the life cycle, i.e. design, development and production. It is shown how the prototype aircraft is defined from the fatigue strength point of view and how flight testing and development testing of the prototype leads in turn to the production definition.

The paper continues with a brief discussion of the "fail safe" and "damage tolerant" concepts. The method of dealing with the fuselage structure itself is next discussed. It is shown that, for the structure, the concern will be to establish an economically viable fuselage under a vibration environment than to establish a safe fatigue life in the true sense.

Finally, after a section on quality control production fatigue testing, a statement of the situation in the U.K. on the in-service digital flight data recording programme is given.

The aims of a programme of fatigue evaluation for a helicopter are to establish that within the working life, the expectation of catastrophic failure from fatigue origins is "extremely remote" and, additionally, the incidence of non-catastrophic cracking is maintained at a sufficiently low level to enable economic utilisation of the aircraft structure to be attained.

These aims have to be achieved within the framework of a minimum weight, minimum cost and maximum performance concept.

This paper provides an outline of the current U.K. approach to this task. The procedure is discussed in general terms, but it should be noted that it has been applied, in its entirety, by Westlands to the Lynx helicopter. It is considered therefore that the practicability of the approach has been well established. Never the less, we must always retain an open mind to the problems involved and any refinements leading to greater realism would be carefully assessed in terms of the benefits gained against the costs of increased testing or calculations.

In the design of the helicopter there are a large number of components where failure could cause structural collapse, loss of control, injury to occupants or cancellation of the mission. These parts are designated Class 1 and are subject to certain special requirements in design and manufacture. However, we in the helicopter industry have gone further than this classification and from the list of Class 1 parts we have selected those parts subjected to fatigue loading whose failure would cause catastrophe. These parts are designated "Vital" and are the subject of very strict design and manufacturing controls.

The main part of this paper is devoted to the substantiation of the safe fatigue lives of these "Vital" parts. I shall make some comments upon the so called "fail safe" and "damage tolerant" approaches later on.

The fatigue substantiation procedure really starts at the design conception of the aircraft and I will divide the whole of the procedure into the following three phases:-

(a) Design phase.
(b) Development phase.
(c) Production phase.

(a) Design Phase (Figure 1)

Figure 1. shows a flow chart of the design activities related to fatigue strength that lead to the definition of the prototype aircraft.

These commence with the operational requirements for the aircraft and lead to the aircraft specification. The specification will define the role utilisation of the aircraft and should provide broad data on the operation of the aircraft in each role. In addition, the specification will call up the various requirements of the airworthiness authorities with which the aircraft is to comply. The specification will also state the minimum fatigue lives required for the aircraft components.

The current airworthiness requirements concerning the fatigue strength of British military helicopters are embodied in the U.K. Ministry Documents Aviation Publication Av.P.970 Vol. 3. In addition, Av.P.970 Vol. 1, (which is generally applicable to fixed wing aircraft) and British Civil Airworthiness Requirements,

Section G, may apply where no general design requirements in Vol. 3, or specific design requirement in the Aircraft Specification exist.

The relevant chapter on fatigue in Av.P.970 Vol. 3 takes the form of a mandatory statement of the fatigue requirements followed by a leaflet containing recommendations of the methods and procedures that may be utilised to satisfy the requirements. However, the sense of the mandatory statement is contained in the opening sentence which requires, simply "that all class 1 parts shall have sufficient fatigue strength for the specified life and operational role". In the recommendatory leaflet fatigue test and flight procedures are suggested and the fatigue scatter factors to be met are stipulated.

Our background of experience at Westlands, up to the time of the Lynx project, had however, led us to evolve philosophies of fatigue substantiation that went much further than the minimum approach featured in the requirements. These new philosophies were, with R.A.E. backing and agreement, put into practice on the Lynx helicopter ard form the substance of this present paper.

However, to continue the discussion of the Design phase of the fatigue evaluation. We derive from the aircraft specification the role utilisation of the aircraft and on this basis a "spectrum of manoeuvres" for each role is obtained. These spectra are of the type given in the American CAM6 and AR56 documents which present the percentage times in various steady level flight conditions and also numbers of manoeuvres per hour.

The next step is to establish the "Design load Spectrum". This spectrum of loads is devised for each component and consists of a simplified pattern of calculated loads considered to cover conservatively all the envisaged role usages of the aircraft. The design load spectrum is used by the stress engineer in his contribution to the design of the prototype part. Preliminary fatigue tests are carried out in many cases and the design modified if necessary to lead to the definition of the prototype aircraft.

(b) Development (Figure 2)

Several prototype aircraft may be built in order to carry out the full development programme of a modern helicopter. No less than twelve Lynx prototypes were utilised for various investigations. These prototypes included a "basic" form of the Lynx and both Army and Navy variants. One of these aircraft was set aside for the load survey and this was straingauged in all critical areas of the dynamic system and fuselage and flown to an elaborate and detailed flight programme at Westlands flight test facility.

The strain gauges are located to measure bending, torsion and total stress at several radial stations on the main and tail rotor blades such that a full flatwise and chordwise stress survey is obtained. Other measurements include loads in the stationary and rotating components of the control system, torque in the transmission shafts, engine input torque, stress on the main and tail rotor hubs and also various critical load paths in the fuselage itself.

The aircraft is then flown to a detailed flight plan to measure loads in steady level flight conditions at all speeds up to about ten per cent beyond the proposed flight envelope limitations. Manoeuvres such as banked turns, pull ups, flares, control inputs and yawed flight are carried out at all appropriate speeds. The rotor speed range, centre of gravity range, all up weight, altitude and various weapon configurations are flown. The measurements of load are analysed, and combined with the manoeuvre spectra, used to produce the production load spectrum for the helicopter in its various roles. The duration of the flight test programme may be of the order of perhaps fifty or sixty hours.

During this development phase a full programme of fatigue testing is being carried out on all vital components. The results of this testing are assessed against the incoming flight load data and the design of the various components is brought gradually towards the production definition.

(c) Production (Figure 3)

The stage has now been reached for the production life substantiation. Components for test are selected at random from the early stages of the production lines. The testing itself is carried out under multi-level programmes of loads which include allowance for the effects of applications and relaxations of centrifugal loads on the rotating components. The load programme is derived from the stresses obtained during development flying. The test aim is to achieve the specified life against a reasonably conservative test programme.

Because of the very large number of cycles of fatigue loadings applied at rotor frequencies to components in the helicopter dynamic system it is our practice to increase the load levels on test well above the flight levels of loading so that a mean test strength is demonstrated that exceeds the so called "Working" strength by an agreed factor.

The results of these multi-level tests are analysed using simple Cumulative Damage theory to establish a "working S/N curve" for each critical section of each component. The production load spectra derived from the development flight programme are then applied to the working S/N curve using cumulative damage theory and the production life substantiated.

As previously mentioned it is necessary to establish a margin between the mean strength demonstrated by test and the strength used for fatigue substantiation purposes. This is required so that the probability of fatigue failure during the service life of the component is "very remote".

There is a difference between the practice of our fixed wing colleagues and ourselves, in the world of helicopters, in determining factors. Critical components of the helicopter dynamic system are usually subjected to loading cycles at rotor order frequencies, rather than gust or manoeuvre frequency, and at the high cycle end of the S/N curve factors upon life become quite meaningless. Our practice therefore is to apply factors to strength rather than life, that is, vertically rather than horizontally on the S/N curve. However, many components in the helicopter are subjected to both high and low frequency loadings,

i.e. main rotor blades under flight bending and on-off centrifugal loads. Other fuselage components may only be subjected to manoeuvre loads. In these cases the low frequency loadings are factored on cycles rather than stress, otherwise the loadings required might take the component into ultimate stress conditions.

Factors have been established for metallic materials in the U.K. on a basis of statistical assessment and experience. The failure probability described by our Civil Airworthiness Authorities (C.A.A.) as "very remote" has been quantified as a failure rate of 1 in 10^7 per hour of flight. It has been postulated that the achievement of this factor on each "Vital" component would still lead to an overall failure probability on the average aircraft fleet that is emotively acceptable.

In order to establish factors to meet this safety level we would require to determine fully the strength distribution curve for the individual components. This is of course quite impractical because of the numbers of specimens required. What we have done in the past however, is to make an analysis of every large sample of identical items that could be retrieved from the literature, or from our own tests. We have tended to find that, in most cases, the distribution appeared to be truncated at the lower end. It seemed therefore that the "Weibull" distribution, which has been said to describe the physical nature of fatigue failure, would be an appropriate tool in the analysis.

On all the samples analysed we found that the use of the three standard deviation level of strength came at or below the zero probability point estimated by the Weibull parameters. This seemed to bear out the experience of ourselves and our American colleagues that such a level would lead to a safe aircraft from the fatigue point of view.

The factors that we are currently using for metallic materials are, in fact, based upon the three standard deviation level and are shown in figure 4.

With regard to composite materials the aim is to establish similar safety levels to those shown by experience to be satisfactory for metals. In the formative stage we are at with regard to composite airworthiness clearance, the aim is to establish scatter data on coupons and structural elements related to the critical areas of the design concerned. This data would enable us to derive scatter factors appropriate to each part concerned allowing working fatigue strengths to be established from the eventual full scale tests. It would be hoped that, in the future, experience will allow us to develop a set of unified factors, similar to those for metals, to be used for design in composites.

Safe Life or "Fail Safe"

The foregoing has provided the briefest outline of the procedures that we in the U.K. are currently using to give fatigue clearance to the Vital Parts of a helicopter. The philosophy used, is termed "Safe Life", and is aimed at the reduction of the probability of initiating a crack by the normal fatigue processes to a negligible level.

However, this procedure cannot guard against the unexpected and abnormal in the fatigue process, and there have been occurrences, fortunately rare, of premature fatigue cracking originating from corrosion, accidental damage in service, damage in manufacture that had escaped the rigorous inspection procedures, abnormal load situations and so on. To guard against such incidents requires new design philosophies. Such a philosophy is the "Fail-Safe" or "Damage Tolerant approach.

It should be noted that there is a distinction between "Fail Safe" and "Damage Tolerant". The former suggests that one cannot be certain that cracks will not initiate at some time during the aircraft life and methods must be devised to detect these cracks before the strength drops below a certain level. Damage Tolerance assumes the existance of initial flaws in the structure. When structure is inspectable then fatigue cracks growing from these flaws must be detected and contained as for the "Fail Safe" philosophy. In uninspectable structure the growth of such a fatigue crack during the whole service life of a structure must be such that a satisfactory level of structure strength is maintained.

For most of the dynamic components of a helicopter the design is such that reliance has still to be placed upon the safe life approach. I believe that one of the major design challenges that we have before us is to apply fail safe or damage tolerant principles to all such "Vital" parts. One part that has already been successfully adapted to these principles is the main rotor blade, on certain conventional helicopters, Messrs. Sikorsky and ourselves have been successfully using a method of pressurising a hollow aluminium spar such that cracks can be detected by loss of pressure shown by an indicator on the blade. This indicator is examined before and after each flight. It has to be demonstrated by test that there is only an extremely remote possibility of a crack starting and propagating to failure within one flight.

Fuselage Fatigue

A major area of the helicopter that has proved difficult to deal with in fatigue terms is the fuselage itself. Certain localised areas of the fuselage i.e. gearbox attachment structure, have simple single load paths and these are treated as any other vital part, i.e. full fatigue tests on local structure full factors etc. The major part of the structure has reasonably fail safe characteristics as it is similar to conventional aircraft build, i.e. semi-monocoque construction of stressed skin, stringers, frames etc. In the initial design phase the structure is generally stressed using the normal static load cases, and, in addition, simple fatigue assessments are made in terms of repeated manoeuvre loading. In service, however, the virtually unpredictable high frequency loadings fed into the fuselage from the rotor system have frequently proved to be a problem requiring local repairs and modifications. These situations are not necessarily dangerous but cause servicing delays and are costly to rectify.

Our practice in the U.K. is to carry out full scale fuselage fatigue tests on the whole fuselage under manoeuvre load conditions. High frequency flight loading is applied at flight load levels to simulate the fretting situation. In this case life factors are used with the manoeuvre loadings. This mode of testing has led with somewhat moderate success to a knowledge of likely problem areas and repair

schemes. It is considered that the solutions for these problems lie in the design phase and there is really no substitute for reduction of vibration at source.

Quality Control Fatigue Tests

One final aspect of production is to ensure that the fatigue characteristics of components on the line do not vary from the standard of early production components from which the fatigue substantiation test specimens were chosen. In order to achieve this it is necessary to select components from the production line at intervals and subject them to a fatigue test. The fatigue test itself may have to be a simplified version of the full production test for time and cost reasons. Components may be chosen on a basis of one in, say, forty production items or once a month whichever comes sooner. The procedure is to choose two components and test one of them. The results of the test are examined against a "Warning limit" and an "Action limit" which we determine statistically from previous testing. Certain parameters are calculated from a comparison of the current result with the results of the previous two tests. If all parameters are within the warning limit, the result is satisfactory, and the second specimen returned to production. If one or more parameters are outside the action limit then the component is defective, production must stop and the cause investigated. If one or more parameters lies between the warning and action limits then the second specimen must be tested. If the results of this are satisfactory then production may continue.

On the Lynx we chose a limited number of vital items for this type of testing including main and tail rotor blades. In general the criteria for selection were based on an assessment of the complexity of both flight loading and manufacture and the likely effect of manufacturing processes on the fatigue strength of the part.

In-Service Load Surveys

There is a thread running through the whole procedure of fatigue substantiation that is fundamental to each phase. This thread is of the utmost importance but is the least well technically understood. I am talking, of course, about the load spectrum. We can fly our prototype helicopter and measure loads quite accurately in all components for a wide range of flight conditions, we can carry out fatigue tests under the most sophisticated load programmes to establish fatigue strength but we do not know completely how the helicopter spends its time in service.

It will be recalled at the beginning of this paper that I suggested that the aircraft specification should provide the broad spectra of operation in each role. To date this has only been provided in a limited way and, in general, has been of a conservative nature. However, the helicopter is a versatile creature and can be used in a wide variety of roles many of which may be devised long after the specification was prepared and the aircraft is in service.

What then is being done about this? In the U.K., one approach we are currently following is an exercise to carry out digital recording on a number of in-service helicopters. The object of this exercise is to record a number of flight parameters over a long period of time to build up a good statistical sample. The parameters will include height, weight, speed, normal acceleration, pitch, roll, heading, control positions, rotor speed, rotor torque and tail rotor torque. Work has already been carried out by R.A.E. using a Scout aircraft showing that a wide range of flight conditions and manoeuvres can be recognised from these recordings. The next step is to relate these manoeuvres to loads in the various dynamic components of the aircraft and this will then lead to the derivation of a full load spectrum definition based on long term service measurements.

Our intention is to equip six Sea King helicopters in service, after a preliminary trial installation, with the digital recording equipment. This can then be expanded to cover any aircraft, i.e. Lynx, when used in any role. It is expected that data will be available from these first Sea Kings in the early 1980's.

We in the U.K. consider that results from this programme will not only enable us to understand the operation of current helicopters much better, leading possibly to improved lives, but also to utilise the information for improving the design input into the next generation of helicopters.

FIGURE 1

D E S I G N   P H A S E

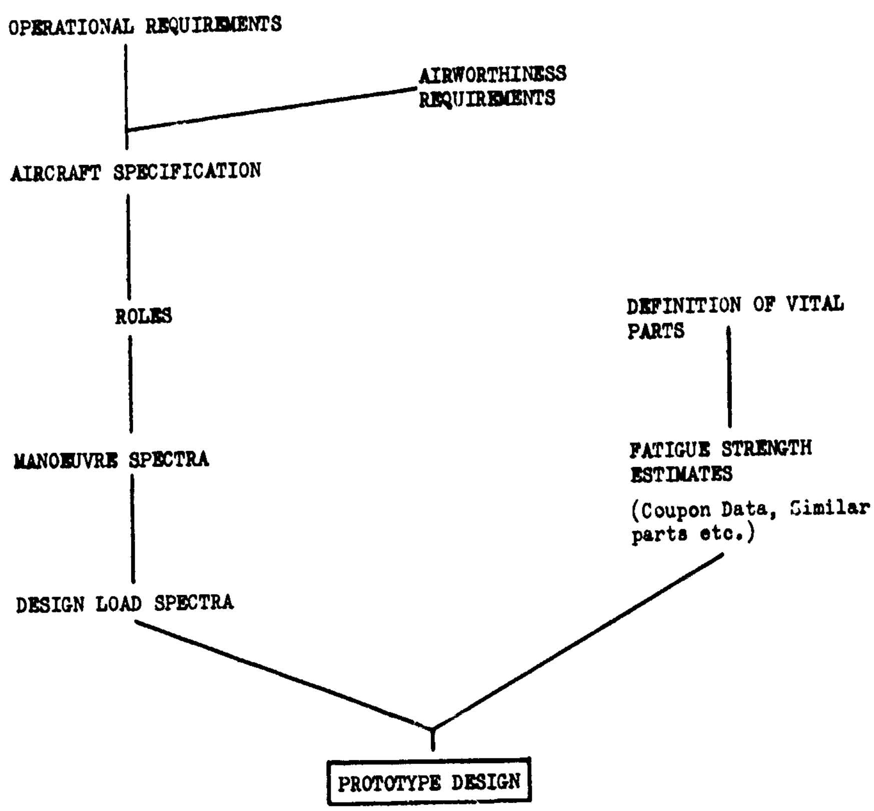

FIGURE 2

DEVELOPMENT PHASE

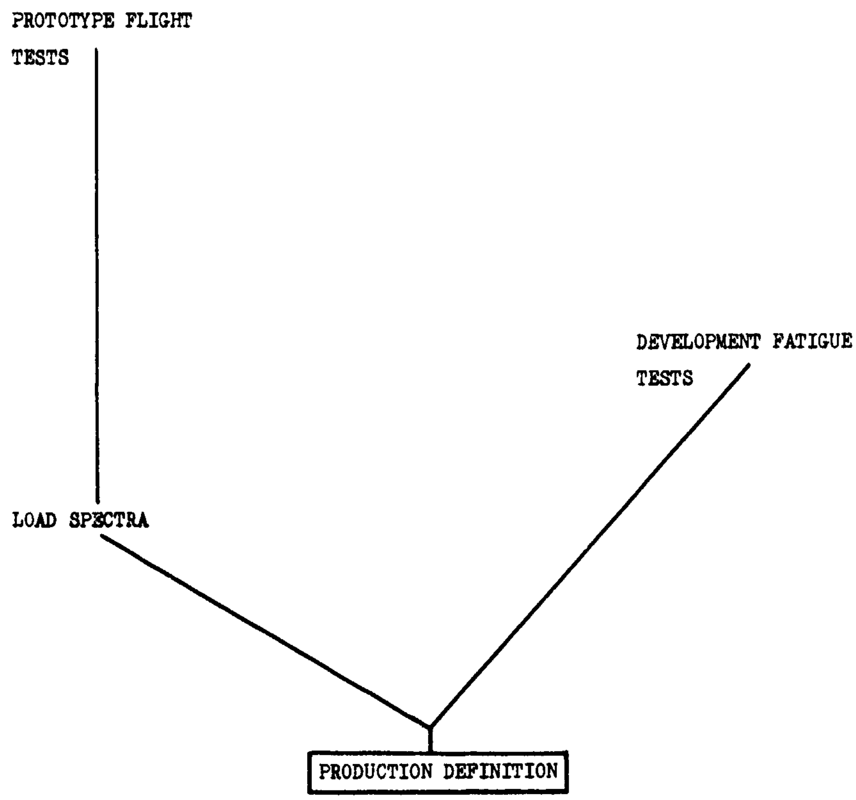

FIGURE 3

PRODUCTION PHASE

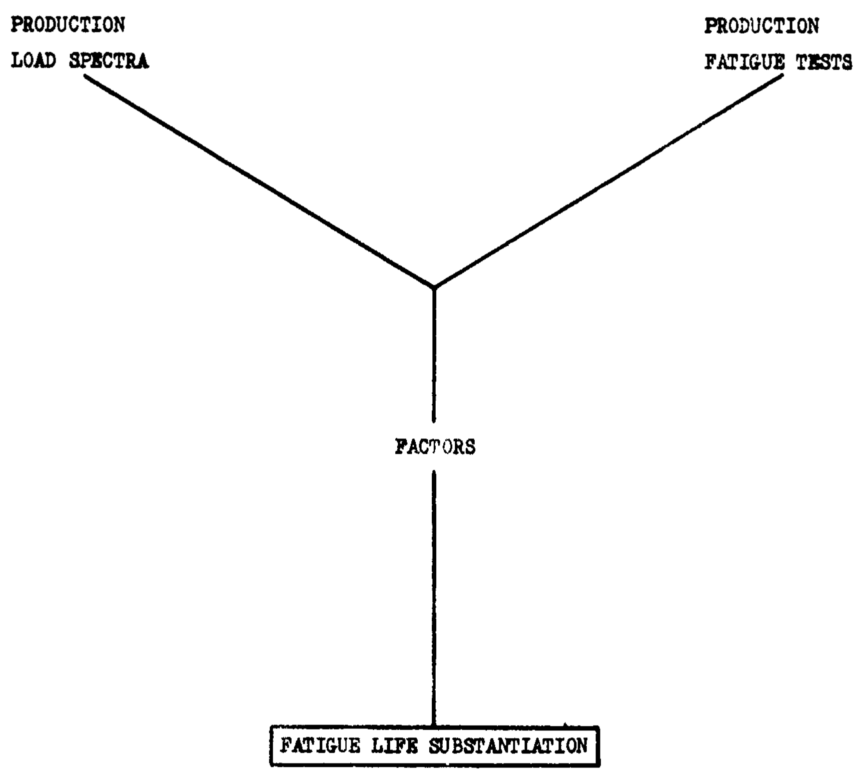

FIGURE 4

F A C T O R S

	FACTOR	
Number of Specimens Tested	Light Alloys	Steel Titanium
1	1.85	1.55
2	1.70	1.46
3	1.65	1.43
4	1.63	1.42
6	1.60	1.40

For Gear Teeth:

4 or more specimens: 1.3
Less than four: 1.4

Flight Scatter Factor: 1.2

FATIGUE LIFE ESTIMATION METHODS FOR HELICOPTER STRUCTURAL PARTS

by

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SUMMARY

This paper gives a short survey on life prediction methods which have been established and proved successful at MBB's helicopter division with primary emphasis on dynamic components.

Analytical fatigue life estimation mainly consists of three steps: prediction of loads, determination of fatigue strength and application of a damage hypothesis linking these two aspects.

Following the above mentioned three steps of fatigue life investigation, this paper will first deal with methods for the prediction of loads according to the available amount of information. Similarly, it shall then investigate methods describing the fatigue strength of components, taking into account the influence of steady loads and the reduction of a mean S/N curve to a working level curve. As to the fatigue damage hypothesis, the "Linear Cumulative Damage Hypothesis", the well known Miner's rule, will be used.

1. INTRODUCTION

It is desirable and necessary to have information on a part's fatigue behaviour already in the early design stage, long before the first hardware is available. In order to avoid unsuitable constructions, the engineer wishes to estimate lifetimes in every stage of design. Naturally, the problem's complexity only allows an evaluation of exact lifetimes for a component when subjecting the full scale part to actual loads. Nevertheless, the mentioned reasons justify efforts in forecasting lifetimes. However, the reliable estimation and prediction of a part's lifetime are still greatly hampered by insufficient knowledge and data concerning the fatigue phenomena and fracture of materials. On the other hand, reliable mission spectra are difficult to set up, especially as operational requirements of light helicopters, such as MBB's BO 105, may vary within a broad field of operation. Thus, fatigue substantiation philosophy, as shown in Figure 1, is used by all helicopter companies with historically based differences in e.g. load application, S/N curves shape criteria, working level curve deviations and component testing techniques [1].

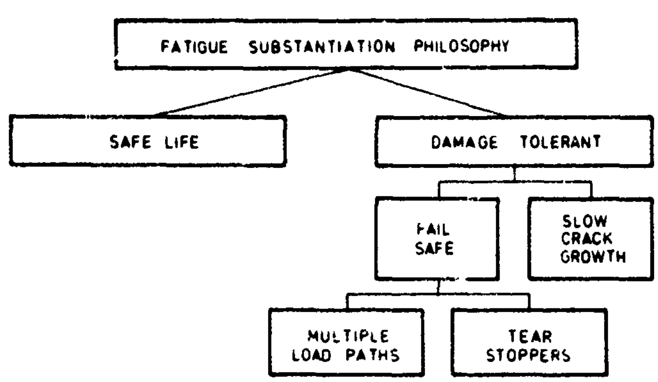


Fig.1 Fatigue Substantiation Philosophy

As almost all fatigue critical dynamic components are safe-life components, the verification of structural reliability with respect to fatigue is a vital task and a reliable fatigue life estimation during all steps of the development is indispensable.

It is the scope of this paper to give a short survey on life prediction methods which have been established and proved successful at MBB's helicopter division with pri-

mary emphasis on dynamic components. Special attention is laid upon the question: which method is best suited at a certain stage of design?

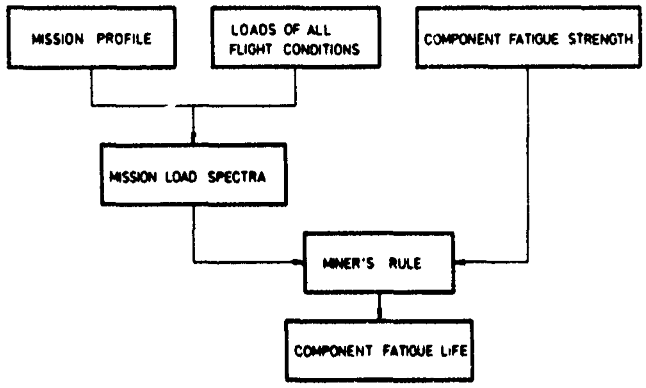


Fig.2 Safe-Life Methodology

As shown in Figure 2, structural fatigue life estimation for helicopter dynamic components consists of three basic steps: (1) the estimation of the magnitude and frequency of occurrence of the loads that will be encountered during the operational life of the structures; (2) the estimation of the components' fatigue strengths taking into account scatter, size and shape effects, surface finish and environmental influences [2]; and (3) the combination of the interactive effects of these loads and strengths by means of a cumulative damage hypothesis.

Following the above mentioned three steps of fatigue life investigation, this paper will first deal with methods for the prediction of loads according to the available amount of information. Similarly, it shall then investigate methods to describe a component's fatigue strength. As to the fatigue damage hypothesis, Miner's rule will be used.

2. LOAD APPLICATION

Mission Spectra

The loads acting on a helicopter's structure are functions of several parameters, e.g. flight conditions, center of gravity position, gross weight, altitude, etc.

The percentage breakdown of all possible flight conditions, known as mission spectrum, is strongly dependent on actual helicopter usage, e.g. civil or military.

For the BO 105, the FAA mission spectrum formed the basis for the determination of the complete load spectrum. Only minor modifications have been adopted. For instance, the high percent occurrence of autorotation was reduced, because the BO 105 is twin-engined. The impact of helicopter mission spectra on fatigue is shown in [3].

Manoeuvre Loads

For simplicity, the case where maximum information is available will be started with, i.e. actual flights have been undertaken and measurements recorded.

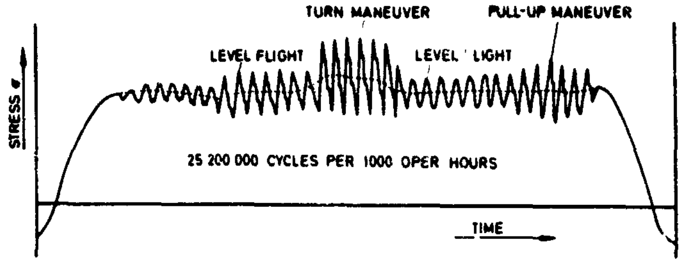


Fig.3 Stress History

An almost always practised simplification reduces the load-time history to cumulative frequency diagrams. Thus, by neglecting the time influence, one ignores frequencies of changing loads and their sequence, As counting method, the so-called "range-pair count method" is used at MBB's helicopter division, resulting in a cumulative frequency diagram, as shown in Figure 4.

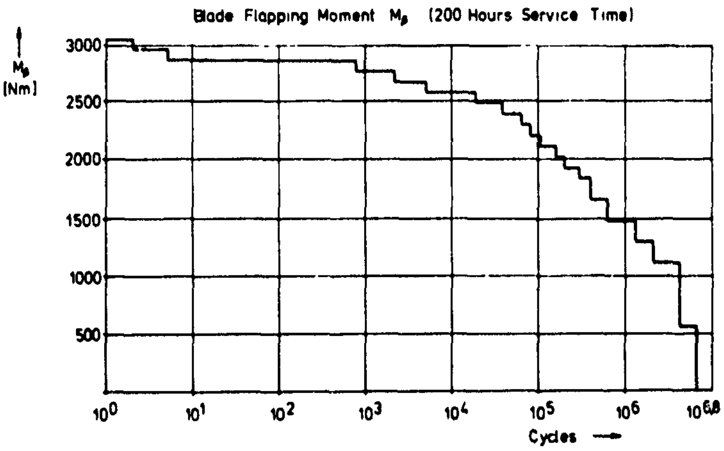


Fig.4 Cumulative Frequency Diagram

In the design phase, flight measurements are not attainable and thus other methods have to be available for the estimation of loads. A simple concept is to assume a similar cumulative frequency diagram as in previous designs for a planned helicopter when similarity of mission profiles, dynamic properties and principles of construction exist. In case these premises do not hold, loads must be calculated assuming simplified structures and using basic mechanical knowledge.

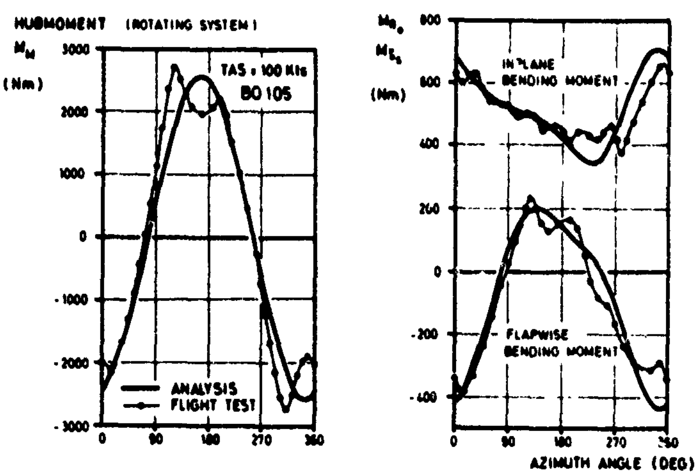


Fig.5 Calculated and Measured Loads

For the hingeless rotor of the BO 105 the first harmonic rotor loads are normally most important and these loads can be computed with relatively high accuracy [4]. There is good correlation between calculated and measured values as shown in Figure 5.

Summing up the frequencies from related flight conditions, e.g. from all turning manoeuvres, and plotting them against the corresponding load levels ends up with the load distribution shown in Figure 6 for the flapwise bending moments acting on the BO 105's main rotorblades during turning manoeuvres.

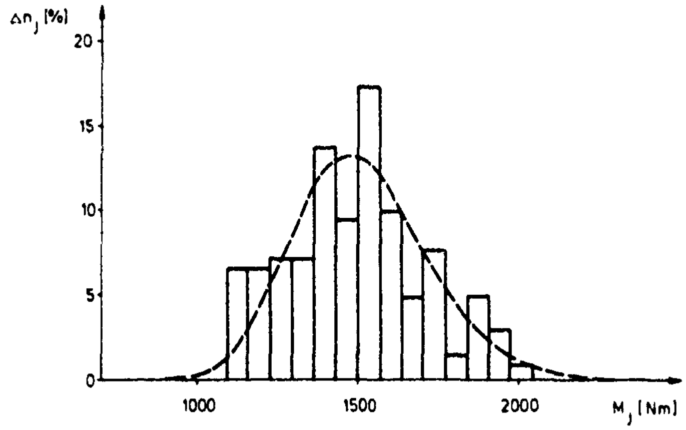


Fig.6 Distribution of Flapwise Bending Moment

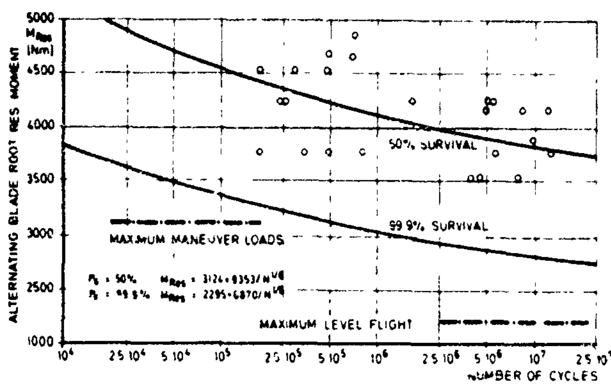
This type of distribution, well-known in physics, economics and biology as the log-normal distribution, is characterized by its median $\tilde{M}$ and standard deviation s and reads

$$\Delta n_j = \frac{(\Delta M_j/\tilde{M})/(\sqrt{2\pi}\cdot s\cdot \ln 10)}{\left(\frac{M_j}{\tilde{M}}\right)^{1+(\log M_j/\tilde{M})/(2s^2 \ln 10)}} \qquad (1)$$

Δn_j being the relative frequency at the j-th load level M_j and ΔM_j the step width. Convenient choice of the lower load boundary results in a constant standard deviation for all flight conditions. This simplification can be justified, as the loads below the endurance limit are assumed to have no damaging effect according to Miner's rule.

3. FATIGUE STRENGTH

Without doubt, the best life predictions will result from tests with actual components and true service loads under realistic environmental conditions. As only at the end of a design and construction process actual components are available, one has to content oneself in the preliminary stages with models or even specimens. On the other hand, realistic loading traces generally contain a high number of cycles with low and medium loads and only a few reversals with high loads. So, in substantiating requested lifetimes of some thousand hours, intolerably long testing times would be necessary. Therefore constand amplitude testing is mostly used to evaluate fatigue strength. Figure 7 shows typical results of constant amplitude bench tests for the root end section of the BO 105's main rotorblade shown in Figure 8.



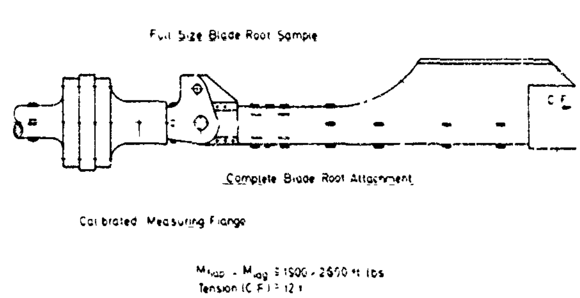


Fig.7 Strength of Blade Root Section

Fig.8 Full-Scale Test Component

Since fatigue strength is of statistical nature, a relatively large number of tests has to be carried out in establishing reliable S/N curves. Statistical information, however, e.g. characteristic shape of the S/N curve and standard deviation, can be drawn from tests of small specimens, as shown in Figure 9, to save money.

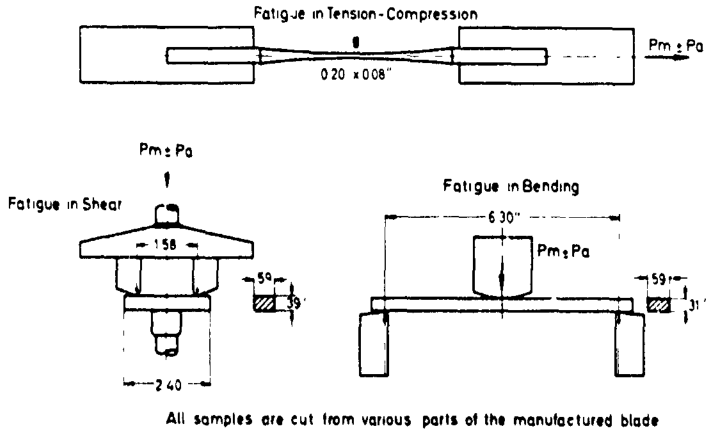


Fig.9 Small Test Specimens for GRP Rotorblades

The simplest method to evaluate a compensating curve from a given scatter band of test results is to draw it by eye. More pretentious, however, is to formulate an analytical expression for S/N curves and to evaluate the free constants by regression analysis. The following simplified Weibull set up has been introduced at MBB's helicopter division.

$$s = s_\infty + \frac{s^* - s_\infty}{N^x} \tag{2}$$

x, s^* and s_∞ are material-dependent parameters, where s_∞ is the endurance limit. Instead of evaluating the exponent x, it may be chosen as 1/2, 1/3 and 1/6 for steel, titanium and glassfibre reinforced epoxy, respectively. The influence of mean load may be taken into account when replacing s^* and s_∞ in Eq. (2) by the following expressions

$$s^*(A) = \frac{s^*}{1 + b/A^z} \tag{3}$$

$$s_\infty(A) = \frac{s_\infty}{1 + a/A^y} \tag{4}$$

A is the ratio of stress amplitude to mean stress and a, b, y and z are material-dependent constants, which must be evaluated by regression analysis.

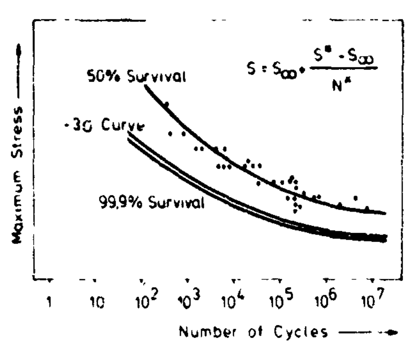


Fig.10 S/N Curve from Test Results

The faired S/N curve represents a survival probability and a confidence factor of 50%. In the helicopter industry, high survival probabilities are required. As a system's survival probability is computed multiplicatively from the single survival probabilities of each component, a survival probability of nearly 100% for the latter is necessary. At MBB's helicopter division, a survival probability of 99.9% is usually used.

4. FATIGUE LIFE ESTIMATION

As already stated, the most reliable fatigue life estimation is obtained when actual components are being tested under realistic loading traces. But in cases neither real components nor realistic loading traces nor both are available, the cumulative damage concept may be used. The standard Miner's rule is required for life determination, based on mission load spectra and working level S/N curves accounting for a probability of occurrence and survivability respectively of 99.9% and a confidence level of 95%.

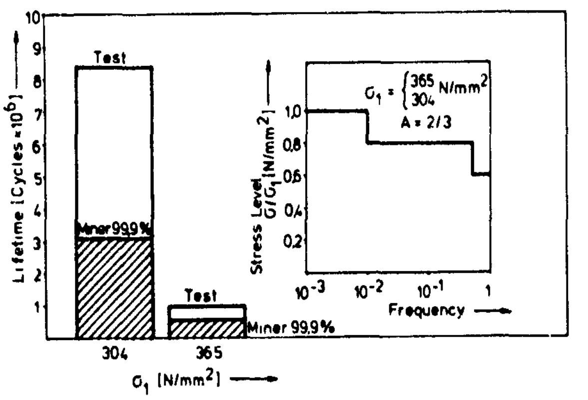


Fig.11 Multistage Laboratory Tests and Analytical Results

It can readily be seen from Figure 11 that the analytically obtained lifetimes lie on the safe side, using a working level curve with a reduction according to a survival probability of 99.9%.

Combining the analytical load formula according to Eq. (1) with the fatigue strength working level curve according to Eq. (2) by means of the standard Miner's rule, one ends up with the following lifetime formula:

$$L = \frac{C^{1/x}/(3600 \cdot f)}{\sum_i \alpha_i A_i \tilde{M}_i^{\,1/x}} \tag{5}$$

with

$$A_i = \sum_j \Delta n_j (M_{ij}/\tilde{M}_i - M_\infty/\tilde{M}_i)^{1/x}, \tag{6}$$

where summation is carried out over all relevant load levels j. C, x and M_∞ are material-dependent parameters, α_i the percentages of the different flight conditions and f the loading frequency. In case mean stresses act, C and M_∞ must be replaced by their reduced values

$$M_\infty(A) = \frac{M_\infty}{1 + \frac{a}{A^y}} \tag{7}$$

$$C\,(A) = \frac{M^*}{1 + \frac{b}{A^z}} - M_\infty(A)\ . \tag{8}$$

5. CONCLUDING REMARKS

In the helicopter industry adequate sizing guarantees that for most parts all loads of level flight conditions and all manoeuvres with frequent occurrence will not contribute to fatigue damage. Only extreme manoeuvres of little occurrence will decide fatigue life. However, as more and more long lifetimes will be required for helicopters, new methods in fatigue substantiation have to be established. According to the random character of load spectrum and fatigue strength the application of statistical means is the only sound method to compute a component's lifetime. The uncertainty in establishing flight load spectra and the problematic nature of the cumulative damage hypothesis have to be compensated by conservative assumptions, as safety cannot be compromised.

6. REFERENCES

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PRESENT FATIGUE ANALYSIS AND DESIGN OF HELICOPTERS REQUIREMENTS AND QUALIFICATION PROCEDURES

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SUMMARY

The-state-of-the-art in AGUSTA in the area of structural fatigue and fail-safe stren gth evaluation is reported. The need of general regulations and procedures is pointed out. The convenience of automatic procedures is underlined.

INTRODUCTION

Fatigue substantiation and determination of service-life are one of the most difficult and important steps involved in helicopters' design. It takes a long time to obtain the results that are necessary either for flight safety or for use convenience; therefore all the problems connected with fatigue must be one of the first steps of design.

Reference n.1 contains a very useful, empirical method to predict fatigue life during a development program. That method allows to update systematically fatigue life estimates. Based on these estimates, any parts indicating a need for redisign are found early in the development cycle and appropriate action can be taken.

It's important to point out that, at the beginning of fatigue tests, flight loads are not yet experimentally determined. Only later they can be correlated to tests results. Anyway, first of all, we must know all requirements to be met by the new helicopter and then use suitable procedures to satisfy these requirements.

Among operational specifications it's necessary to define those connected with fatigue problems. For example, for a military helicopter it's necessary today to consider either requirements connected with components' operational life or requirements typical of Damage Tolerance as to withstand war damage.

Finally, safe-life fatigue evaluation is, at present, only an aspect of fatigue Design. We must assure helicopter's safety also in presence of damages considered up to now not repairable and catastrophic.

FATIGUE REQUIREMENTS

Fatigue phenomenon is articulated into two different phases :

1) Nucleation : until the beginning of crack.
2) Propagation: until failure

Everyone of these phases needs proper methodology analysis and tests. For this reason there are, at present, the two following methodologies:

a) Safe-life Methodology
b) Damage-Tolerance Methodology

Design requirements of a civil helicopter, based on Safe-life and Damage-Tolerance approach intend to assure :

A) Service safety in the assigned Spectrum
B) Competitive fatigue life limits and inspection intervals.

These requirements are codified into the most common Civil Regulations as reference n° 2 and n° 3.

Specific regulations are added to military helicopters, such as:

C) Strength to war damage
D) Slow crack propagation rate if damage is accepted.

Generally detailed requirements for principal components are :

- Safe-life components must have a minimum of 5000 hours of operational life.
- Critical primary structure will be designed to withstand design limit load with projectile damage without failure.
- Critical primary structure will be designed to sustain projectile damage for a minimum of 30 minutes of flight.

It is also requested to minimize the probability of catastrophic failure due to propagation of undetected flaws, cracks or other damage.

- No structural components must have less than 30 flight minutes from time of detection of damage by a failure warning system to complete failure.
- Components impractical to be visually inspected and not provided with failure warning systems must be considered safe-life.

PROCEDURES TO MEET FATIGUE STRENGTH REQUIREMENTS OF MAIN STRUCTURES.

The most important civil and military requirements for the helicopter Design have been analyzed. These Requirements are subjected to changes, additions or cuts with reference to the different types of helicopters; but, when defined, they must be entirely met. The principal objective of designers is to satisfy these requirements by proper structural design, reliable methods of analysis and suitable tests.

We have already mentioned two typical procedures of Fatigue Design: the Safe- life Methodology and the Damage Tolerance Methodology.

Requirements to be met by civil helicopters are codified in Federal Aviation Regulations (FAA-FAR, PART 27 and 29) and in Civil Air Regulations (CAA-BCAR, G3). Unfortunately sometimes the procedure to comply Requirements is not clear. For example chapter 571 of FAR 27 and 29 (Reference n. 2) defining Requirements to be met in fatigue evaluation doesn't refer to the appendix A of C.A.M. 6 (Reference n.5) which was the guide for the fatigue substantiation procedure during Fifties and sixties years.

English Regulations, on the contrary, report either Requirements or Procedures to meet Requirements.

For military helicopters, finally, is often necessary to adopt reliable procedures or agree with National Control Authority about new ones.

Only for last military helicopters there are accurate indication about the methodologies to use in order to comply requirements. However these indications are very specific and often can not be generalised.

MISSION SPECTRUM

One of the most important items for fatigue evaluation of critical structural components is the mission spectrum. This subject was already discussed by AGARD (Reference n.6)

Nevertheless its importance calls for another little discussion because mission spectrum is the source of the loads spectrum.

Sometimes helicopter's rôle is not definable from the beginning of Design.

We can just define the field of use: civil or military

Civil helicopters have a lot of use : executive, cargo, off-shore, research and rescue, cargohook, agricultural operations, etc.

If they are multi-purpose, the definition of mission spectrum must be very general.

Civil Regulations suggest percentages of various flight conditions in order to give a guide to the Designer. These percentages can be modified in relation either to some helicopter's pecularities (single-engine, twin-engine) or to the prevalent use or to the operational experience of similar machines.

A typical civil spectrum includes a lot of flight conditions and few high load factor maneuvers.

Military helicopter's purpose, on the contrary, is today very specific. Mission spectra are established from the beginning of Design. Typical of these spectra is a lot of high load factor maneuvers.

To compare civil and military spectra is just impossible; anyway it appears clear that the military ones are heavier (high percentage of maneuvers, high load factors, a larger number of take-off and landings) due to the use near the ground and to the higher manoeuvrability.

In those cases in which helicopter's rôle is established the mission spectrum is known and the only problem is to use one of the two fatigue evaluation methods. It is however possible, especially for civil helicopters, to consider some other usages and also the military one.

At this point is necessary to analyse again the design spectrum and evaluate again main components.

For same components it will be necessary a new design. In other cases components' retirement lives will be reduced when new design is not economical. Generally speaking, we can say:

1) Military helicopters' spectra are very heavy and their civil usage extension is enough simple.
2) Civil helicopters' spectra need, on the contrary, a new analysis in order to meet military requirements.

Only if fatigue evaluation is computerized, helicopters' manufacturers can increase the number of missions also after some years from the delivering, in accordance with customer's requests.

Agusta can analyse every safe-life component in a little time, using a computer program. The only thing to do is to change the percentages of occurrence of different flight conditions in order to obtain the new operational life.

Appendix A show the operational mission spectrum utilized for the A 109 A's fatigue evaluation.

SAFE-LIFE METHODOLOGY FOR CRITICAL COMPONENTS.

During last years AGUSTA's objective has been to improve and computerize safe-life fatigue evaluation.

Detailed discussion about the methodology of AGUSTA can be found in Reference n.4.

Here is reported a summary of that work in order to show that fatigue life calculation procedure becomes automatic and simple utilizing a fast data acquisition System and an adequate calculation methodology.

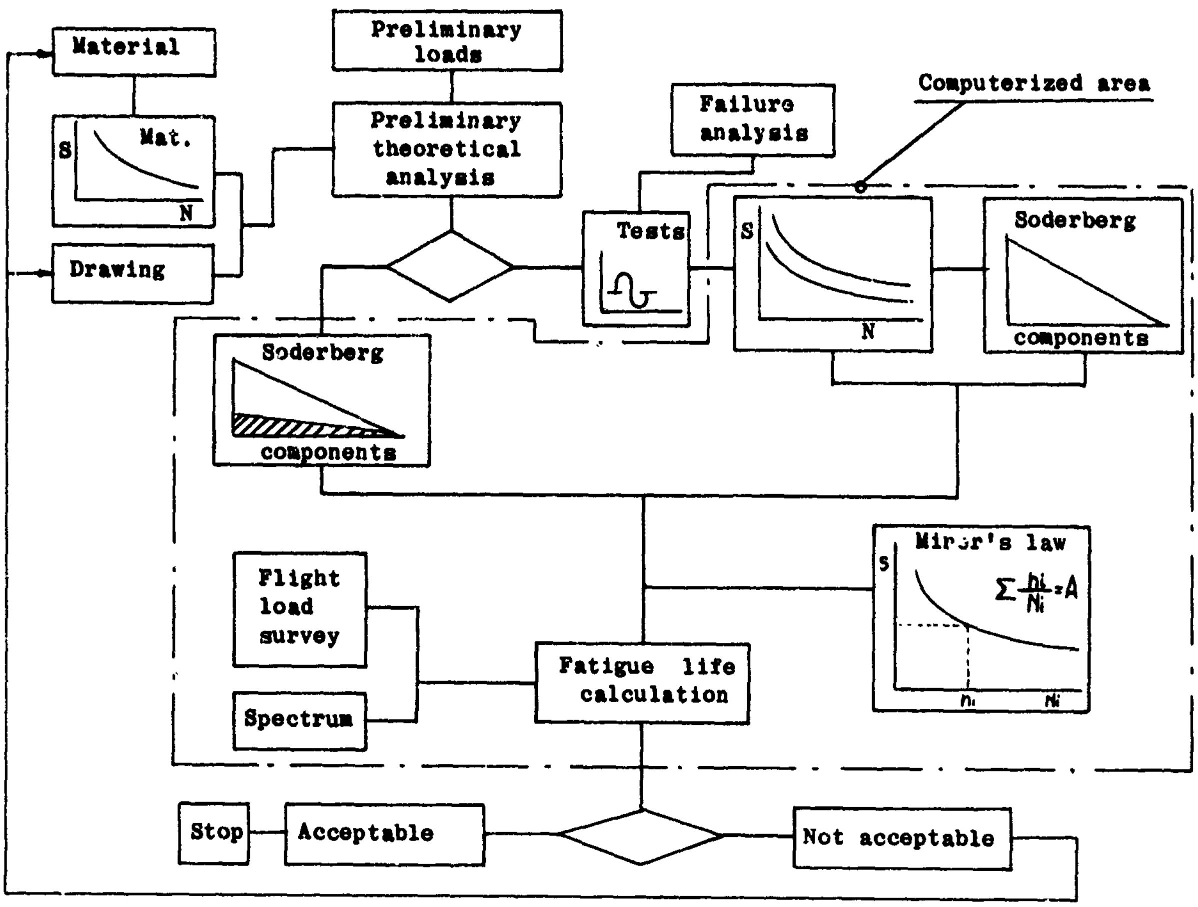
Fatigue qualification is articulated into the following steps :

1) Materials' selection
2) Materials' Wohler's curves.
3) Operative component Wohler's curves
4) Soderberg's Diagram.
5) Constant Amplitude Fatigue tests.
6) Examination of failure areas.
7) Flight load survey.
8) Mission Spectrum
9) Load survey data elaboration and component fatigue life calculation

Figure n° 1 relates to this procedure.

The dot-and-dash line contains the computerized steps.

Principal features of each step and their use are now explained



Fatigue qualification's procedure

Figure 1

1) Materials' selection.
It is important to get high values of the strenght/weight ratio and good properties with reference to stress corrosion, notch sensitivity, etc...
A lot of materials are available on the market with their properties so the selection process is possible with reference to previous requirements.

2) Materials' Wohler's curves.
Usually Weibull'· definition is adopted:

$$N = \frac{K}{\left(S - \frac{E_\infty}{E}\right)^M}$$

The determination of constants $K, M, E_\infty/E$ can be accompl ·hed in two ways:

A) Using results of tests on specimens of the same material of component.
B) Using results reported in literature.

AGUSTA defined the values of $K, M, E_\infty/E$ reported in table 1, taking into account four materials classes : steel, alimunium alloys, titanium alloys, and magnesium alloys.
These values are on the safe-side and can be modified according with particular design or detailed data.

Material	K	M	E_∞/E
Steel	6.4×10^4	1.76	.94
Aluminium alloys	4.76×10^5	3.322	.80
Titanium alloys	$1. \times 10^4$	1.969	.97
Magnesium alloys	$1. \times 10^5$	7.4	.6

TABLE 1

3) Components' operative Wohler's curves. In another paragraph we shall investigate problems involved with fatigue tests. Now we suppose to have carried out tests so we can draw for every component Wohler's curve, using Weibull's equation.
The curve must pass over failure points. Four tests are necessary, at least, for each component.

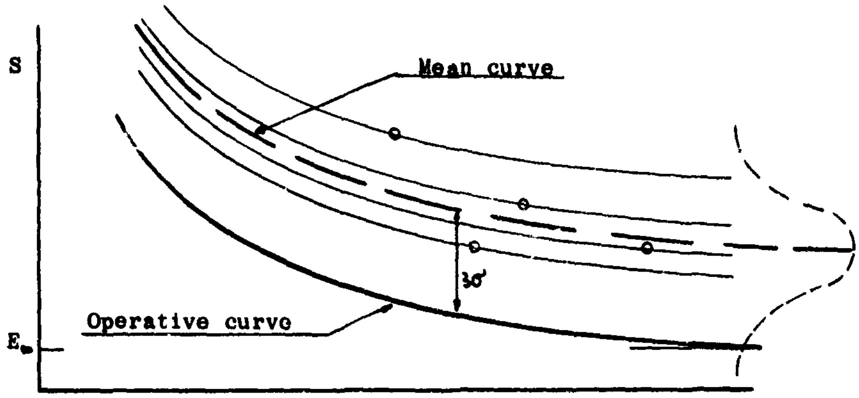


Figure 2

Operative Wohler's curve

Afterwards, under the assumption of a Gaussian distribution of values at $N=\infty$, we obtain the mean curve. This curve is reduced of 3σ , the new curve is the operational curve of examined component.
If 3σ reduction is lower than 25% for components of aluminium alloy or lower than 20% for component of steel, the mean curve is reduced of these percentages.
Anyway calculation is accomplished using either the load or the logarithm of load.
The greatest value of reduction defines operative curve of component.

4) Soderberg's Diagram.
When the component's operational curve is found, it is possible to define its Soderberg's Diagram.

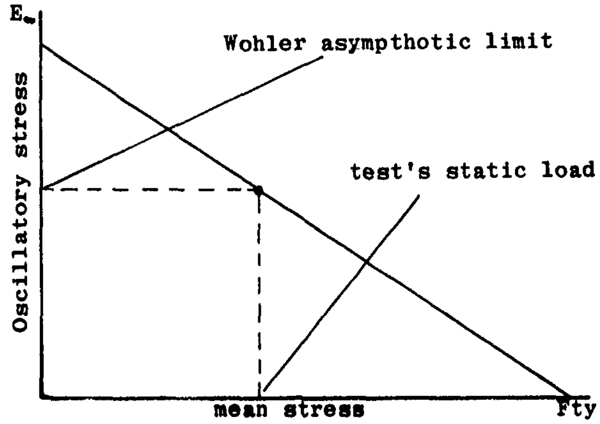


Figure 3

Experimental Soderberg's diagram

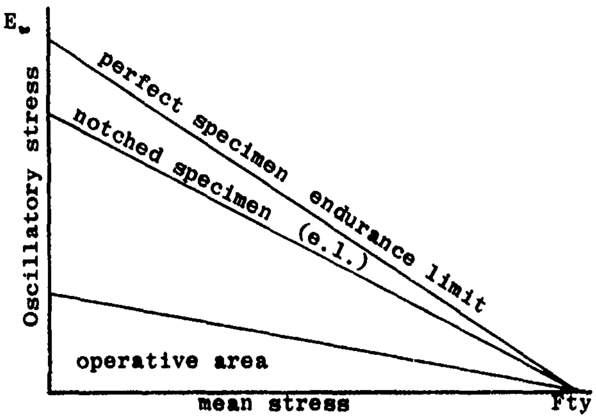


Figure 4

Analitical Soderberg's diagram

This diagram allows to correlate flight data with test data because it allows to determine the endurance limit for every flight mean stress also different from test mean stress.

We can obtain the same diagram also by mean of theoretical approach (without fatigue tests) for those components subjected to very low stresses whose fatigue safety is large.

In this case Soderberg's operational diagram of components can be obtained from Soderberg's operational diagram of material by mean of a first reduction of K_f and of a second reduction of [illegible].

If flight stresses fall into the last area no specific fatigue tests are necessary.

Some considerations on the procedure are :

a) Soderberg curves have on X-AXIS, as a limitation, the yield point and they are assimilated to straight lines.

b) The first reduction would not refer to the theoretical stress concentration factor K_T but to the fatigue strenght reduction factor wich is usually equal to or less than the theoretical stress concentration factor, except in cases where fretting is involved, however, K_T is more frequently used.

c) The importance of a correct evaluation of K_T is evident, especially when combined stresses are present.

d) The flight data are expressed in terms of net stresses on the critical areas of each component.

e) Caution should be exercised in the application of this procedure when the following items are involved :

e.1.) Parts subject to fretting
e.2.) Bolted or pinned connections
e.3.) Irregulary shaped parts containing numerous holes, threads or lugs.

Now we want to underline on aspect wich is becoming more and more important.

By the finite element analysis it's today possible to determine the local stress value near high stress concentration zones, with a great precision.

In these cases the K_T reduction should be omitted.

Reference n.7 describes an important experience of AGUSTA about this subject.

5) Fatigue tests.

Fatigue critical components can be divided into two groups by mean of a theoretical analysis : components requiring fatigue tests and components not requiring fatigue tests.

This way to operate involves some risk due to the fact that analysis loads are not flight loads which are not yet available.

This risk can be minimized using available data of similar helicopters.

Appendix B contains a summary either of A 109 A's main components subjected to fatigue tests or of main components only analytically checked.

With reference to components requiring fatigue tests, these tests has been constant amplitude tests carried out by universal machines (small components) or by other machines (big components). See photo pag. 6-7

6) Examinations of failure areas

The test is considered completed when the failure of the components intervenes or when it reaches, under load, the number of cycle typical of "infinite life".

Photo 1

Fatigue test of main rotor servo control

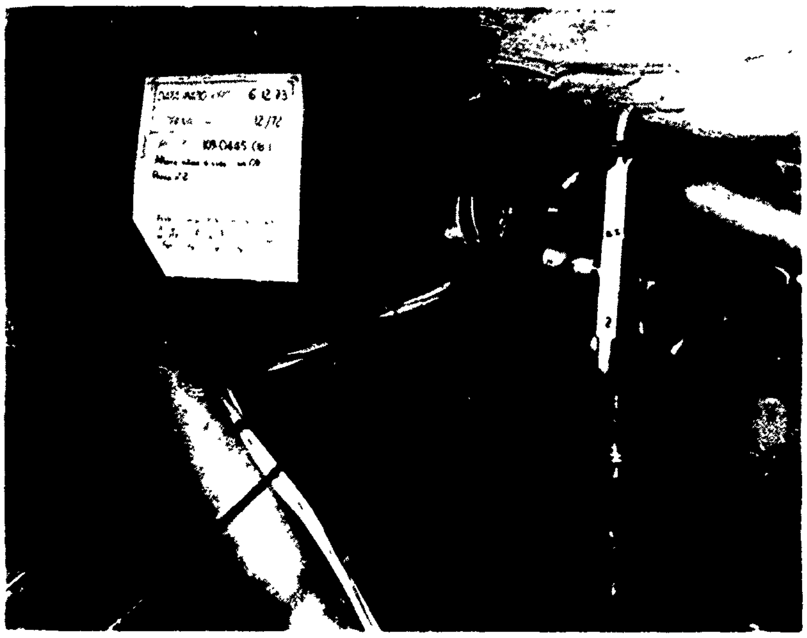


Photo 2

Fatigue test of tail rotor mast

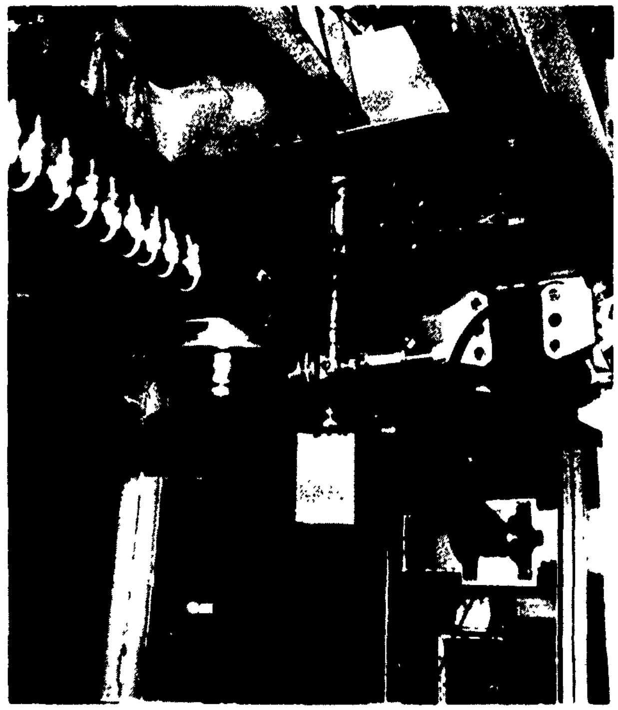


Photo 3

Fatigue test of main rotor mast

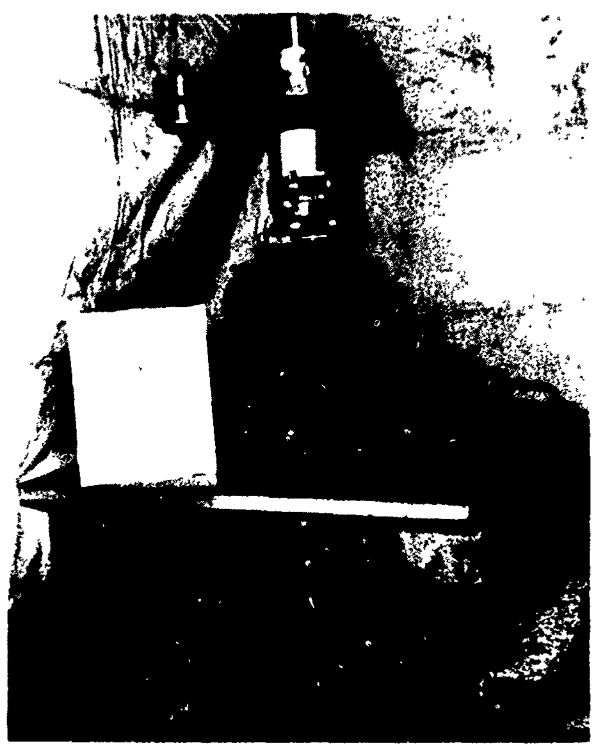


Photo 4

Fatigue test of swashplate support

Not destructive examinations (optical, microhardness, penetrant dyes, magnetic particles) are performed on the most significant sections, critical areas, bonded areas, where the failure does not appear clearly, to point out possible incipient failures or superficial damages. If the examination indicates some failure, next point was applied.
On the failure section, a morphological examination of macroscopic type is made in order to determine :

- Origin's zone of the failure and the cause which produced the failure.
  If the origin is on the surface, as usual, the condition of the surface around the failure section is analyzed.
- Material's homogeneity and isotropy
- Defects, if present, in the failure section, as blows, shrinking holes, large or iso-oriental crystallization, segregation, non metallic inclusions.
  If the macrographic examination showes the existence of one of such defects, the micrographic investigation is utilized to prove its existence.
- Characteristics of the area of the fatigue failure.
  When the area is not damaged excessively we make :
  - striations marking to have indications about the material's grain
  - examination of striations propagation to have the crack propagation speed magnitude.
  - Examination of the simmetry of the area where striations lie, with respect to the crack starting point and to planes involved.
    It is an index of the nature of the stress which caused the failure.
- Characteristics of static crack propagation area.
  Each examination is represented in a laboratory report with photographic documentation.

7) Flight Load Survey
Flight load survey can be divided into four parts :

7.1) Helicopter's preparation
7.2) Flight plans' preparation
7.3) Execution of flights
7.4) Data acquisition

7.1) Helicopter's preparation
First of all it's necessary to identify critical zones of each component.
Later on strain-gauges shall be put in these zones.
Calibration and connection to board recorder are last steps.

7.2) Flight plans' preparation.
These phase's aim is to organise flights in order to reduce time. Therefore spectrum conditions are divided with reference to loads, centre of gravity, altitude, etc.

7.3) Execution of flights.
Flights execution begins after preparation. Critical fatigue components' data acquisition must be quick in order to supply the necessary feed-back to theoretical analysis.

7.4) Data acquisition
Technique FM/FM (Frequency modulation/ Frequency multi-plexing) is used for data acquisition.
Signal from strain-gauges are conditioned and modulated on different subcarriers (CBW or PBW, IRIG standard).
Carriers obtained in this way are either recorded on a magnetic support (WB recorder with 14 tracks, group 2) with 100 parameters capacity or transmitted to ground (link in L band) with a capacity of 21 parameters.
This process is explained in figure n. 5
Ground System (S.A.N.D.R.A.) collects such data, transforms the carriers and returns signals in their original shape.
These data, through a minicomputer's monitoring, are digitalised for every parameter at 1000 samples for second rate, coded in binary form and reduced on a digital tape (9 tracks, 1600 BPI) compatible with every computer.
This process is reported in figure n. 6

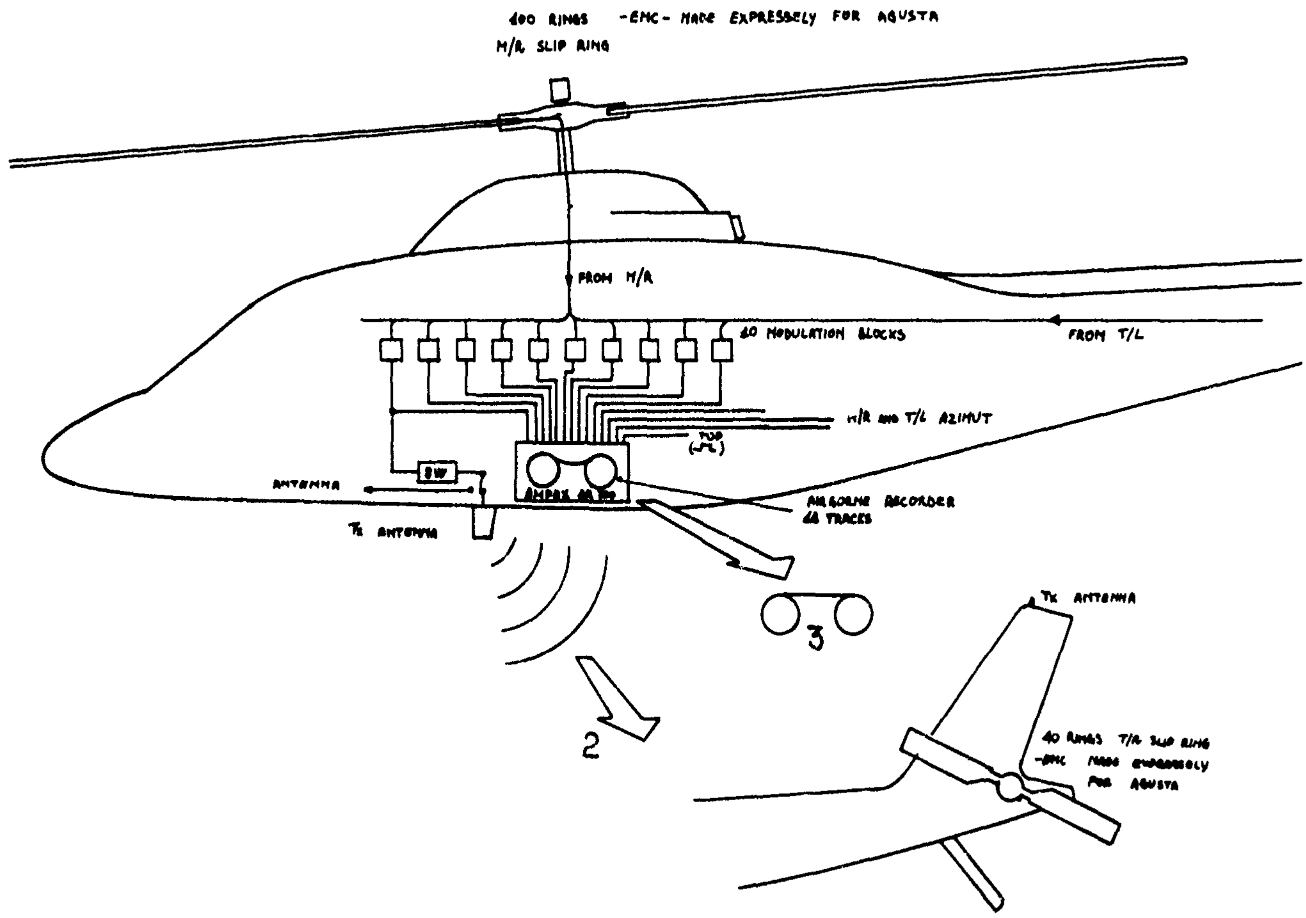


Figure 5

Flight load survey data acquisition

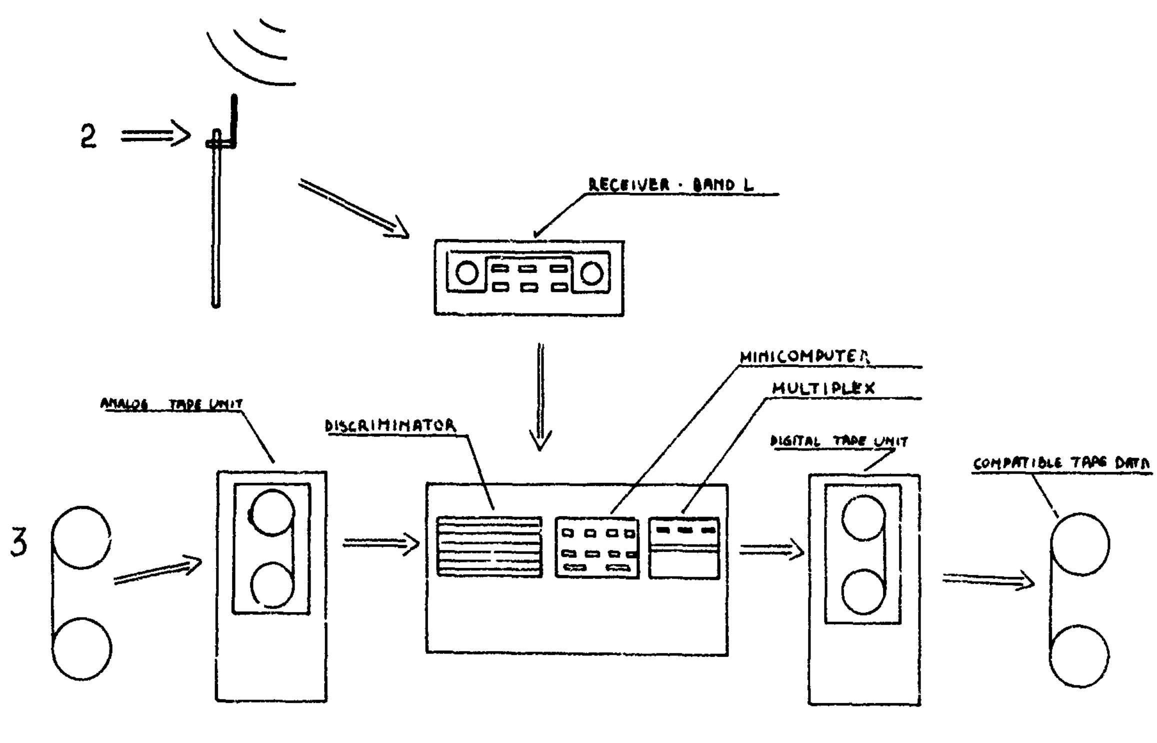


Figure 6

Flighi load survey data elaboration

8) Mission Spectrum
Appendix A reports the A 109 A's mission spectrum

9) Load survey data elaboration and components' fatigue life calculation.
Data processing outlined in figure n°5 pag. 9 shall be explained afterwards.
Now we want to point out that it's possible the calculation of components' fatigue life using Wohler's curves, Soderberg's diagrams and Load Survey Data.
It's however necessary to define a cumulative damage rule :

$$\sum \frac{n_i}{N_i} = A$$

We use Miner's rule and so A is 1. The automatic calculation's procedure is showed in figure n° 7

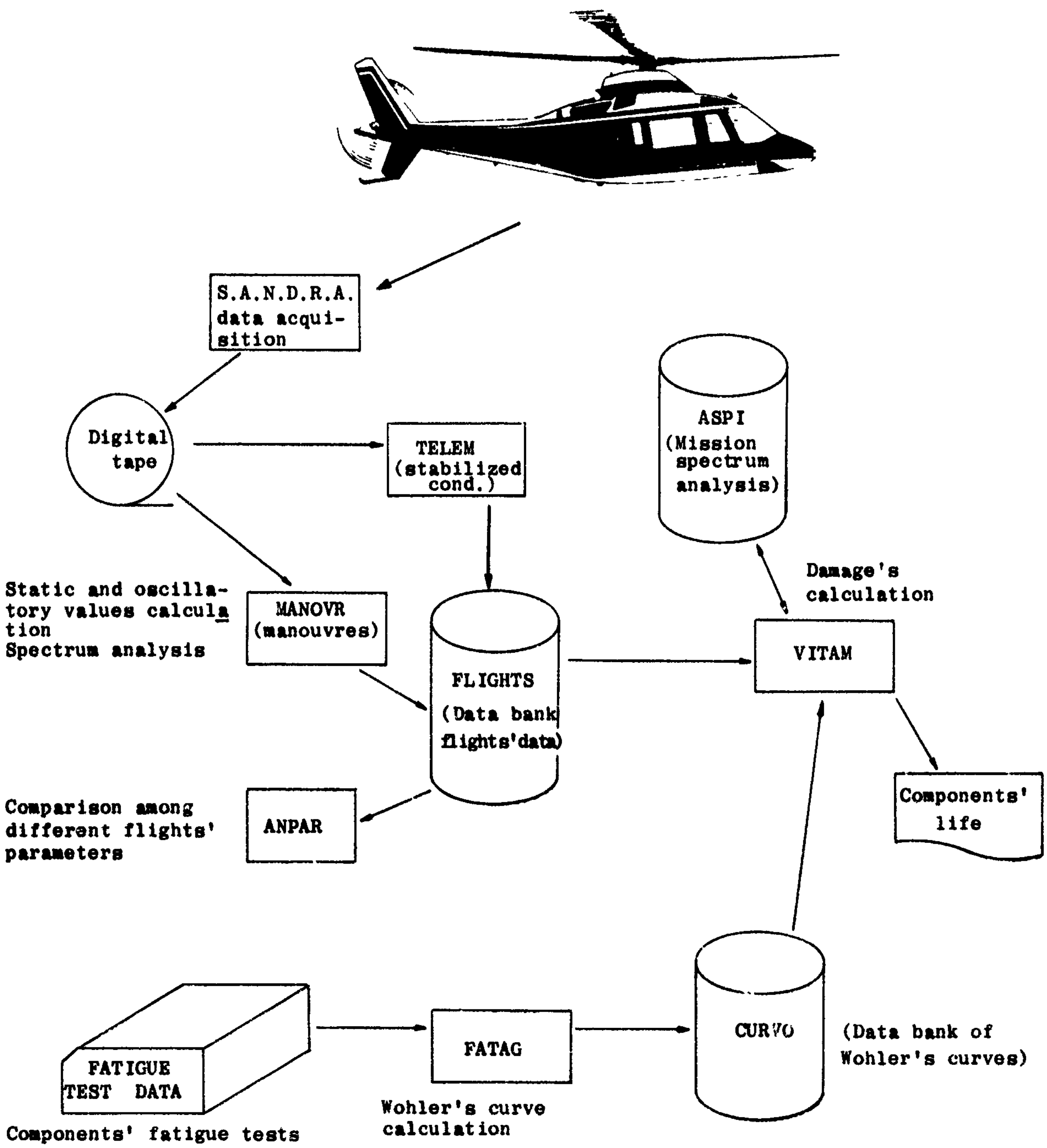


Figure 7

Automatic calculation of components' life

Digital tape from Flight Load Survey is elaborated by two programs in order to obtain components' stresses
One program takes into account stabilized conditions, the other one maneuvers.

- TELEM Program (stabilized conditions).
  The signal refering to the parameter is divided into some parts as long as time of one main rotor's revolution and then is examined.
  Every revolution the minimum, the mean and the maximum value of static and alternatic parameter are calculated.
  The analysis goes on for a maximum of 15 revolutions (about 2.5 seconds), then we determine minima and maxima values and at last we calculate mean values.
  These values are stored on a file (Flight Data Bank File) with those parameters necessary to identify them (flight number, condition, etc.) See figure n° 7 pag. 10

- MANOVR Program (maneuvers conditions) operates in the same way with reference to maneuvers.
  The only difference is the definition of the manoeuvre's time about wich we shall talk later. From "Data Bank" File is possible, by means of AMPAR program, the plotting visualization of each flight condition's loads.
  At the same time fatigue tests data are elaborated by FATAG program and introduced into CURVO File in order to obtain the Wohler's curve of components. Another file named ASPI, contains the informations concerning mission spectrum (flight conditions, percentage of occurence, etc.)
  An important step for following calculation of damage and component's operative life is the determination of alternating load's frequency.
  Usually helicopter's components are subjected to the vibrations of rotors and these vibrations are not similar to sinusoidal waves.
  We transform these vibrations into a sinusoidal wave with the amplitude given by the largest random amplitude, the mean value given by the previons measured value and the frequency equal to the weighted mean of all random frequencies.
  At this point Vitam program can operate in order to calculate components' fatigue life.
  Let's give a more detailed look at the main elements of this process.

- FATAG : in figure n.8 is illustrated how this program works.

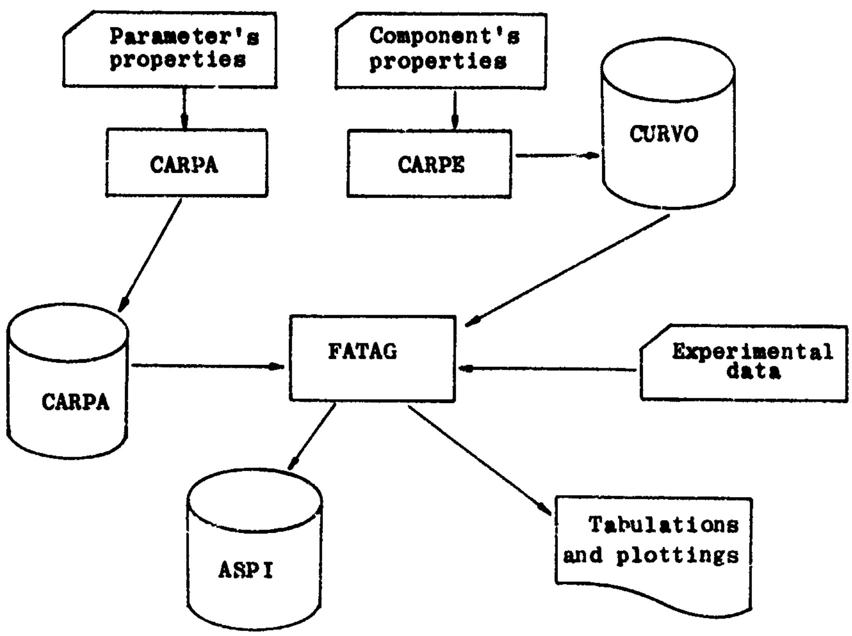


Figure 8

"Fatag" program

- CARPE program loads onto the data set CURVO component's properties and associates components and parameters. Fatag program reads component's experimental data from cards, component's properties from CURVO, instrumentated parameter's identification item from CARPA.
  Then Fatag updates CURVO by limit load and experimental mean load, informs Aspi

that Wobler's curve has been generated and gives plottings of Wobler's curve.

- VITAM the objective of this program is the determination of components' life starting from flight tests data obtained in accordance with mission spectrum.
Vitam Calculates and stores on Data Set Aspi the component's damage for all flight conditions.
For every condition it's possible to take into account up to 150 components. After calculation, Vitam comulates either elementary damages or times of flight conditions.
Component's life is then :

$$\text{LIFE} = \frac{\sum \text{time}}{\sum \text{damage}}$$

Figure n.9 pag. 13 explains life's analytical calculation.
In order to understand figure n.9 it's necessary to give some definitions :

Period time :"TPER"
"TPER" is the time for one main rotor revolution. This time determines the length of each analysis of the signal for both stabilized flight conditions and maneuvers conditions and includes the signals from both the main and tail rotors.

Condition time : "TC"
For stabilized conditions, as the signal running repeats itself periodically in the time, the "TC" was defined as 9 segnal periods.
The procedure for determining "TC" for flight maneuvers is more complex.
First, all of the signal from the entire recording time which included all of the complete maneuver, is analyzed for damage values. They are calculated considering the elementary periods as an aggregate of stabilized conditions.
We consider 4 different maneuver time definition:

a) "TC" is the time equal to the summation of all damage periods.
b) "TC" is the time between the beginning of the first damage period to the end of the last damage period.
c) "TC" is established on the basis of previously conducted tests.
d) "TC" is the recorded tape conversion time, established manually by the pilot who evaluated the manoeuvre performance.
In other words, the pilot started the recording at the manoeuvre initiation and stopped it at the manoeuvre completion.

The choice of one of these definitions and its applicability is a function of the conservativnes we want to obtain. The definition in a above is the most conservative, progressing to d.

Static and alternating flight signals: C_{SV} C_{AV}
For each considered condition, C_{SV} and C_{AV} are referred to the period. There are 9 static and 9 alternating values for the stabilized conditions in the 9 elaborated periods and as many as the periods of the tape recording are, for the manoeuvres.

FEQ :
A constant level value for the alternating load which is clearly lower than the infinite life value is assumed.
When the flight alternating load exceeds that assumed value, during either stabilized or maneuver conditions the signal spectrum analysis automatically starts. This occurs within every period. The result of this analysis is "n" amplitudes and "n" associated frequencies.
Let "A_1" the amplitude of the frequency "F_1", "A_n" the amplitude of the frequency "F_n", with "F_1"< "F_2" < ...<"F_n". If $P_i = A_i/A_1$ and $Q_i = F_i/F_1$ with "i" from 2 to "n", during the fatigue life calculation the signal is considered equ[illegible] to a signal of the same value of the static and alternating load but with the fre[illegible]ncy equal to the equivalent frequency ,(FEQ)which, for definition, is a function of "P_i" and "Q_i".
This is made for each period.

The flight cycles "n" in the period is now determined by multiplyng the equivalent frequency by "TPER", period time. At this point we define C_{SP} as the mean load in test and C_∞ as the asymptotic load value.
A static value correction is now made utilizing the Soderberg diagram for each period, when the flight static value is not equal to the static value.
A new reduced or increased infinite life limit is now found for all of the components.
After determining the new infinite life limit, the Wohler diagram is entered with the maximum flight alternating load to meet the curve to the failure cycles number'N"

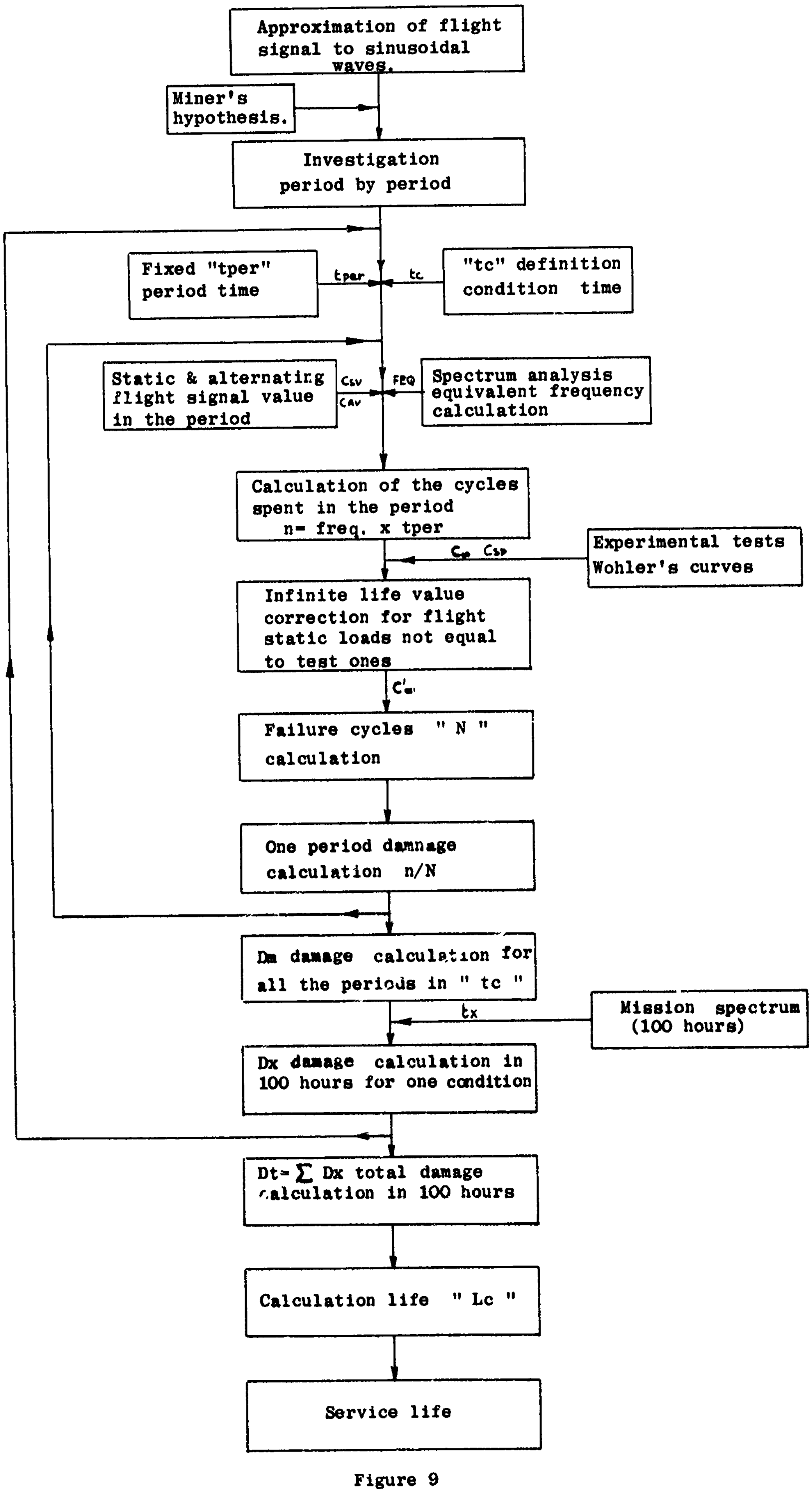


Figure 9

Flow chart of life's calculation

This is made for each period and for each condition of the mission spectrum. Of course, if such maximum alternating load is less than the new limit, the resulting damnage is "0".
The total damnage D_T for 100 flight hours on all components is determined by the summation of all spectrum damnage, made condition by condition and period by period.
The component life L_C for a damage equal to 1 is then calculated.
For this calculation there were 2 considerations :

- If the component is not subjected to ground-air-ground (G.A.G.) cycles or if the relative damage to a component which is subjected to G.A.G. cycles is equal to "0", then :

$$L_C : 1 = 100 : D_T$$

$$L_C = \frac{100}{D_T} \quad \text{(hours)}$$

- For the components which are subjected to G.A.G. cycles and suffer damage the life L'_C is first calculated, considering only the damage which results from the flight phases.
  Then :

$$L'_C = \frac{100}{D_T} \quad \text{(hours)}$$

After L'_C is known it is possible to calculate the G.A.G. cycles number (4 each hours) spent in L'_C hours and the relative damage D_{PA}.
L''_C is then calculated in the second approximation:

$$L''_C = \frac{100}{(D_T + D_{PA})} \quad \text{(hours)}$$

Repeating the procedure, D''_{PA} is determined until :

$$L_c^i - L_c^{i+1} < 1 \text{ hour}$$

The calculatd life is equal to L_c^{i+1} (hours) rounded to the safe side whole number.
The service life of the components, according to Regulations, is now obtained in the following ways :

- If calculated life $<$ 3350 hours
  Service life = 0.75 x calculated life
- If caluclated life $>$ 3350 hours
  Service life = 0.375 x calculated life + 1250 hours

At last we want to point out that the Component's Retirement life can be lower than Service life for some reason indipendent of fatigue evaluation.

At the end of processing can be necessary to modifie mission spectrum. New life of components is then calculated storing the new spectrum and using previous, "Flight Bank" and "Curves Bank".
If new experimental data are available, new life can be calculated changing only Wohler's curve.
Vitam program realize the change and calculate the new damage.

DAMAGE TOLERANCE PHISOLOPHY COMPONENTS.

The basic assumption of DT philosophy is that cracks are always present in structures. The objective of the damage tolerance approach is to design the structure in such a manner that, should damage occur, the structure would retain adequate residual strength until the damage is detected and repaired.

It goes without saying that the designer must develop an inpection program based on analytical methods, tests and previous operational and design experience.

In this way it will be possible to assume as minimum life of components the safe inspection interval. Afterwords, if cracks are not present or detectable, service life of the component shall be prolonged of another safe inspection interval.

Accordingly to this approach the designer must select materials, geometry of structures and service-stress in order to have low rates of crack propagation and large critical sizes.

Redundant structures are often used in order to slow down or arrest the crack growth transferring loads from cracked to safe components.

Structures that have the capability to withstand specified loads after fatigue failure of a principal element are said FAIL-SAFE.

In order to evaluate safe inspection intervals are commonly used the principales of Fracture Mechanics and Nondestructive Inspections.

With reference to AGUSTA is alredy began a development program whose objective is to define a computerized methodology suitable for FAIL-SAFE Design of some critical components.

The basis of such methodology is a critical investigation of those methods which allow statistical analysis and computation of flight data.

Next step is a critical investigation of existing methodologies in the field of variable amplitude fatigue analysis. Damage tolerance approach begins at this point as an aspect of Facture Mechanics.

Last step shall allow to define a calculation program whose output are ispection intervals of critical components, cracks' critical lenghts and everything needs in Damage Tolerance Design.

This methodology shall be support by accurate nondestructive control techniques (especially with reference to flaws' detection) and by a large number of experimental tests.

CONCLUSIONS

At last we want to mention again the most important points examined :

1) The need to unify procedures necessary to meet requirements.
2) The importance to computerize analytical methodologies.
3) The utility to define mission spectra as general as possible.
4) The consciousness that future structures either Safe-Life or Fail-Safe must withstand for short periods large cracks.

APPENDIX A

A 109 A'S MISSION SPECTRUM

The normal operational spectrum includes 8 basic flight phases :
- Ground conditions (engine start, taxi, shut down).
- Take-off (normal, acceleration to climb air-speed)
- Stationary flight (forward, lateral, rearward, turns, hovering, longitudinal control reversal, lateral control reversal, rudder control reversal - in ground effect and out of ground effect-)
- Climb (twin engine, one engine inoperative (O.E.I.) 2 to 1 engine transition).
- Level fligth (forward to 1.1. VNE, turns, cyclic and collective pull up at .6 and .9 VNE, longitudinal, lateral and rudder control reversal at .9 VNE, acceleration from climb air speed to .9 VNE, deceleration from .9 VNE to climb air speed, 2 to 1 power transition at .6 and .9 VNE, 2 engines to autorotation power transition at .6 and .9 VNE) (Twin engine, O.E.I.)
- Autorotation (normal, turns, flare, auto to 2 engines power transition).
- Descent (normal - twin engine, O.E.I.-, 1 to 2 engines power transition)
- Landing (approaching and normal landing-twin engine, O.E.I.-)

Each phase is then subdivided into various subconditions (58) in accordance with the numerous combinations of flight parameters: gross weght, c.g. location, altitude, rotor r.p.m.

The final values of the percentage endurance for each flight condition and expected operation is established after a statistical study, which assigned a probability percentage to the variables ; weight distribution, number of passengers, quantity of luggage and fuel. The flight survey either verifies the proposed values or offers suggestion for their prudent modification.

This is to say that if a flight condition, featured by a combination of parameters values proved critical for the loads, the condition percentage is increased with respect to the assigned one.

We assigne the following probabilities to the 9 zones of the CG/weight envelope, (see fig. A1)

- High weight
  Total percentage: 30%

  25% FWD CG : (7.5% of the total one)
  65% NEUT CG : (19.5% of the total one)
  10% AFT CG : (3.0% of the total one)

- Normal weight
  Total percentage : 50%

  15% FWD CG :(7.5% of the total one)
  70% NEUT CG :(35. % of the total one)
  15% AFT CG :(7.5% of the total one

- Low weight
  Total percentage : 20%

  5 % FWD CG :(1% of the total one)
  75% NEUT CG :(15% of the total one)
  20% AFT CG :(4% of tho total one)

It can be seen, from fig.A1 , how the flight load survey investigates the most critical points of the CG/weight envelope.

We choose 3 level for weight, CG and altitude and 2 level for r.p.m. in the load survey execution.

APPENDIX B

FATIGUE CRITICAL COMPONENTS

We define "fatigue critical component" as one subjected to alternating flight loads whose failure can be catastrophic for the safety of the flight and of the passengers.

The following groups of components were subjected to an experimental evaluation:

Main Rotor hub and blade

- Blade
- Grip assembly
- Retention strap assembly
- Spindle

- Hub assembly

Main Rotor flight and elevator controls

- Rotating scissors assy
- Rotating swashplate
- M.R. pitch control rod
- Stationary swashplate
- Swashplate support sleeve
- Stationary scissors assy
- M.R. hydraulic servo

Tail Rotor hub and blade

- Blade
- Retention strap assembly
- Hub assembly

Tail Rotor flight controls

- T.R. pitch control rod
- T.R. equilizer
- T.R. hydraulic servo

Transmission

- M.R. mast
- T.R. mast
- FWD and AFT pylon support rods
- Servocontrols to transmission cover attachments
- Transmission upper case

Structure

- Elevator assembly

The most important ones among the components only analitically evaluated are :

- Engine fitting
- Upper and lower fin
- Landing gear

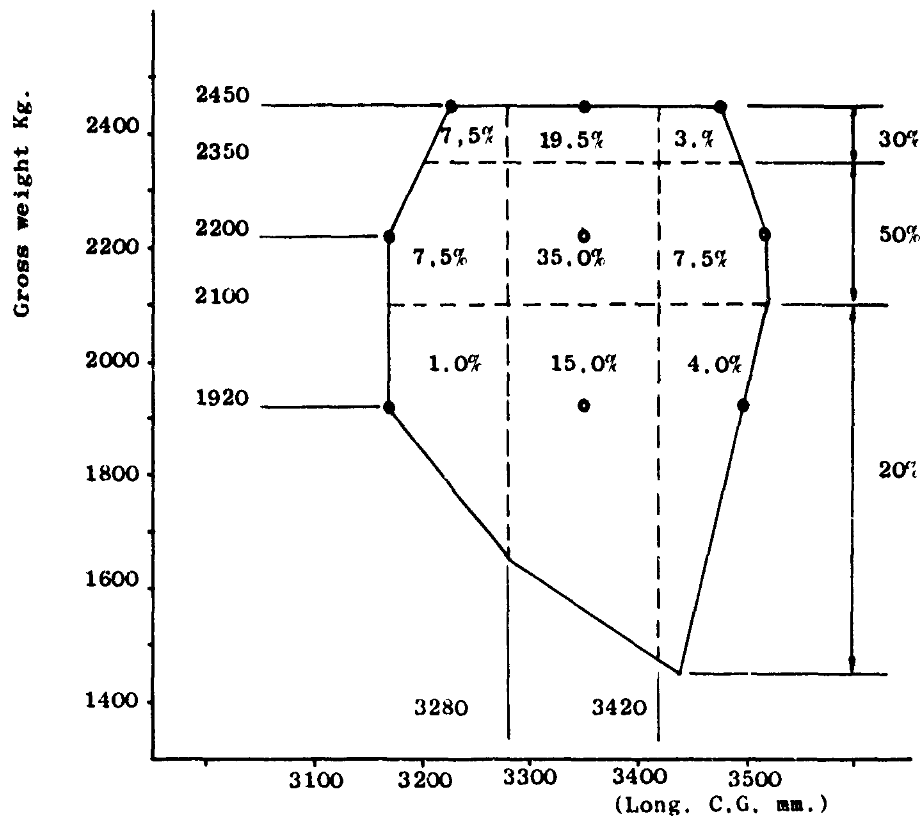


Figure 1A

Weight and balance percentages

• Points investigated during load-survey

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FATIGUE DES HELICOPTERES - METHODE D'EVALUATION DES DUREES DE VIE

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1 - INTRODUCTION :

Les méthodes de justification à la fatigue des pièces d'hélicoptères n'ont guère évolué depuis une dizaine d'années à quelques raffinements près dus au progrès des méthodes de mesure, à l'accumulation des essais et à l'expérience acquise en service.

D'une manière générale, on évalue la résistance à la fatigue de l'élément considéré, on détermine les charges auxquelles celui-ci sera soumis en service ainsi que leur fréquence d'application puis, en utilisant ces données, on fixe les mesures à prendre pour que le risque d'accident grave dû à la défaillance de cet élément soit très reculé :

- aucune mesure (résistance très surabondante)
- retrait de service à un certain nombre d'heures (limité ou non)
- périodicité d'examen pour recherche de détérioration
- combinaison des deux limitations précédentes

Toutes les méthodes utilisées aujourd'hui sont basées sur le fait que la résistance à la fatigue est une propriété aléatoire dont la distribution est sensiblement gaussienne à condition de choisir une variable statistique convenable.

De plus, à de rares exceptions près, on cherche à couvrir les appareils dont l'utilisation est la plus sévère, compte tenu des fréquents changements d'usage qu'un même hélicoptère est amené à effectuer.

Actuellement, on traite de manière différente les pièces essentiellement dimensionnées par la fatigue à grand nombre de cycles (rotors et réducteurs) et celles qui sont soumises à la fatigue oligocyclique (fuselages par exemple).

La sécurité est prise sur la contrainte dans le 1er cas et sur le nombre de cycles dans le 2ème.

Lorsque les deux types de fatigue interviennent, différentes méthodes permettent de tenir compte de leur superposition, mais on rencontre des difficultés pratiques dans chacune d'entre elles.

Il semble logique, puisqu'il s'agit d'un même phénomène, de prendre une approche unique pour ces deux modes de sollicitations en se fixant le risque global consenti, soit que l'on calcule le dommage à chaque niveau de charge, soit que l'on conduise les essais sous charges programmées. Dans les deux cas, il faut connaître les courbes iso probabilités de rupture entre quelques cycles et l'infini, ce qui n'est pas réalisé aujourd'hui.

On a maintenant démontré que les matériaux composites étaient susceptibles du même traitement que les métaux en ce qui concerne la tenue à la fatigue et que l'on pouvait bénéficier du caractère "fail safe" lorsque les premiers dommages apparaissent à l'extérieur. Les stratifiés permettent de réaliser des structures comparativement très performantes en masse ou en durée, mais en contrepartie une surveillance très stricte de la qualité doit être exercée si l'on veut éviter des variations importantes de caractéristiques de fatigue dues à la complexité d'élaboration de ces matériaux.

2 - METHODES ACTUELLES POUR LES PIECES CONVENTIONNELLES :

La durée de vie d'une pièce est fonction d'une part des charges auxquelles elle est soumise en service et d'autre part de sa résistance à la fatigue. Au cours du temps, nous avons cherché à serrer de plus près la réalité dans l'un et l'autre de ces domaines pour arriver à la solution la plus économique possible - moindre poids et plus grande longévité - tout en conservant un niveau de sécurité acceptable.

2.1 - Charges en service :

C'est sans conteste dans ce domaine que l'effort a été le plus marqué.

D'une part, on ne s'est plus contenté des estimations des spectres de vol par des commissions composées des contructeurs, des utilisateurs et des services officiels, mais on est passé à leur mesure effective en service.
La première opération de ce genre, lancée par les services officiels français pour le PUMA et coûteuse en apparence, a été extrêmement rentable puisqu'elle a permis de doubler la durée de vie des pales et de tripler pratiquement celles d'éléments importants de commande comme le plateau cyclique.

A titre d'exemple, la fig. 1 montre la différence entre la répartition des facteurs de charges estimés et mesurés : on voit que ceux ci sont mieux répartis que prévu en fonction de la vitesse et que la majeure partie d'entre eux correspond à des virages de moins de 30° d'inclinaison.

D'autre part, la mécanisation des méthodes de dépouillement a permis de remplacer l'effort enveloppe de chaque configuration utilisé auparavant par un histogramme de charges basé sur l'analyse du signal enregistré tout au long du cas de vol considéré.

En ce qui concerne les manoeuvres dont la répétitivité n'est pas bonne, ce travail doit être effectué un nombre de fois suffisant pour représenter correctement le vol moyen.

Là encore, le gain sur les durées de vie est très important comme le laisse penser la distribution des charges dynamiques sur une pale en flare représentant la moyenne sur 100 manoeuvres de ce type (fig. 2).

Il est évident que toute nouvelle méthode de calcul dans laquelle intervient la sévérité d'utilisation de l'appareil doit être sanctionnée par l'expérience. On peut donc se demander si les "raffinements" ci dessus ne diminuent pas de manière dangereuse la marge de sécurité pratique inhérente aux anciennes méthodes.

En fait, l'expérience (nombre d'heures de vol) acquise depuis 10 ans avec cette méthode est supérieure à celle dont on disposait précédemment et ceci sans que l'on ait pu trouver d'accidents dûs à une marge insuffisante en fatigue. De plus la meilleure connaissance des charges en vol conduit dans certains cas à diminuer la durée de vie des pièces, ce qui va dans le sens de la sécurité.

On remarque enfin que le type d'utilisation a une influence marquée sur les durées de vie. La fig. 3 donne, à titre d'exemple, les durées de vie selon mission d'un hélicoptère de moyen tonnage. Celles ci sont représentées par la longueur des segments ombrés, la durée de vie en transport étant prise pour unité.

Il apparait donc financièrement intéressant de donner des durées de vie variables avec le type d'utilisation.

Cela ne se fait que pour les différentes versions d'une machine et dans la mesure où les pièces correspondantes ne sont pas mélangées au cours des révisions générales : c'est le cas des versions civiles et militaires.

On admet aussi de fixer un coefficient d'utilisation à appliquer aux heures réelles de vol pour quelques pièces importantes et dans des cas où certaines machines sont réservées à une mission particulière (pulvérisation agricole ou ASM par exemple). L'examen de la fig. 3 montre aussi que la masse de l'appareil a une influence prépondérante sur la durée de vie de la majorité de ses composants. C'est donc l'étude statistique approfondie de la répartition des masses en utilisation qui serait la plus payante.

2.2 - Résistance à la fatigue et courbe sûre :

Il est maintenant bien connu que la résistance à la fatigue est une propriété aléatoire que l'on peut représenter, pour une pièce donnée, sous forme de courbes S.N.P (contrainte cu charge, nombre de cycles, probabilité). Si le réseau de courbes correspondant est connu (fig. 4) et si l'on se place dans le cas simple où les sollicitations de la pièce sont à niveau de charge constant, on peut facilement déterminer le nombre de cycles (ou d'heures de vol) auquel il convient de retirer la pièce pour ne pas dépasser un risque de rupture donné à la fin des vols.

La courbe iso-probabilité correspondant à ce risque est la courbe sûre ou courbe de travail.

La méthode utilisée varie selon que la pièce en cause est principalement soumise à la fatigue vibratoire (charges faibles et grand nombre de cycles) ou la fatigue oligocyclique (charges élevées et faible nombre de cycles).

2.2.1 - Fatigue vibratoire :

2.2.1.1-Equation des courbes de fatigue :

Les très nombreux essais de fatigue disponibles montrent que, pour les nombres de cycles supérieurs à 10^5 - domaine auquel nous limitons la fatigue vibratoire -, les courbes S/N iso probabilité de rupture peuvent être représentées par une équation type unique ne dépendant que du matériau et de la présence ou del'absence de corrosion de contact.

Selon que le matériau possède ou ne possède pas de limite de fatigue, cette équation est de la forme :

$$S/S_{\infty p} = 1 + \frac{A}{N^{\gamma}}$$

ou :

$$S/(S_{N=1})_p = \frac{1}{N^{\gamma}}$$

expression dans lesquelles S_{∞} p et $(S_{N=1})$ p représentent la limite de fatigue et la résistance à 1 cycle (ou à 1 mégacycle) ayant la probabilité I-P d'être atteinte ou dépassée par la famille de pièces considérée.

En d'autres termes, P est la probabilité d'avoir une pièce dont la résistance à la fatigue est inférieure à S_{∞} p ou $(S\ N = 1)_p$ selon le cas.

Remarquons en passant que l'hypothèse d'affinité des courbes iso probabilité de rupture formulée plus haut ne peut être vérifiée qu'aux probabilités élevées (1/10 ou à la rigueur 1/100) faute d'avoir un nombre suffisant d'essais.

Le tableau de la fig. 5 donne les coefficients des courbes S/N que nous utilisons pour les matériaux les plus usuels. Ces derniers sont déterminés par essai de 100 éprouvettes au moins dont la forme est adaptée au matériau et le dépouillement est effectué par régression linéaire en coordonnées logarithmiques.

2.2.1.2-Choix de la variable statistique :

La forme de la courbe S/N étant celle du paragraphe précédent, il convient de déterminer la loi de distribution de la limite de fatigue S_{∞}(ou de la charge de rupture par fatigue à un nombre de cycle donné S_N) pour pouvoir extrapoler les valeurs expérimentales jusqu'à une probabilité suffisamment faible pour être acceptée en service.

Pour celà, on recherche une variable statistique, fonction de S ou de N,qui soit gaussienne dans le domaine expérimenté.

Remarquons tout d'abord que, par nature du phénomène de fatigue, cette distribution est limitée aux valeurs positives des deux coordonnées et que S ou N ne conviennent pas. Log S ou log N donnent de bons résultats aux valeurs élevées de S (fig. 6) mais ce n'est pas le cas pour log. N aux faibles valeurs de la contrainte S (moyenne et écart type infinis).

Comme nous nous intéressons spécialement aux contraintes faibles dans le cas de la fatigue vibratoire, c'est donc log. S_{∞} que nous choisissons comme variable statistique et la fig. 7, tirée du contrôle statistique à la fatigue des pales d'Alouette III, montre que cette hypothèse est bien vérifiée par l'expérience.

2.2.1.3 - Courbe de fatigue sûre :

Celle-ci doit être déterminée à l'aide des essais de fatigue et nécessite tout d'abord la connaissance de la limite de fatigue moyenne et de l'écart type de la pièce considérée.

Les travaux d'une commission d'étude dirigée par le Service Technique Aéronautique français et portant sur l'analyse de très nombreux résultats d'essais sur pales d'Alouettes ont conduit aux conclusions suivantes :

- la moyenne de la famille est bien définie à partir de six points d'essais (à ± 5 % près environ pour une dispersion usuelle)

- un très grand nombre d'essais est nécessaire pour estimer correctement l'écart type.

En conséquence, la limite de fatigue moyenne est déterminée à partir d'essais de pièces réelles mais l'écart type doit être fixé par l'expérience antérieure sur une famille importante de pièces semblables.

Dans le cas des pales, il a été établi que cet écart type était voisin de 0,06 pour les alliages légers et de 0,045 pour les aciers et ces valeurs ont été généralisées à l'ensemble des pièces de l'appareil.

La moyenne et l'écart type de l'échantillon étant connus; il convient de choisir un risque acceptable et d'en déduire le facteur de dispersion définissant l'affinité entre la courbe moyenne et la courbe sûre.

Le calcul est effectué,est basé sur la méthode de M. SOULEZ-LARIVIERE dans laquelle on admet un risque de rupture égal à 10^{6} avec une loi de probabilité tronquée pour annuler la probabilité de rupture aux environs du tiers de la limite de fatigue.

Le nombre d'écarts types de l'échantillon à porter sous la courbe moyenne est calculé à partir du nombre d'essais disponibles en utilisant la formule de HALD tenant compte des erreurs sur la moyenne et l'écart type à un degré de confiance de 90 %, suivant fig. 8

La limite de fatigue sûre ainsi obtenue, exprimée en % de la limite moyenne, est alors corrigée suivant fig. 9

Cette troncature de la loi de probabilité correspond à l'usage général n'exigeant pas de justification expérimentale lorsque l'effort maximal de vol est inférieur au tiers de la limite de fatigue. Elle correspond au fait que le contrôle de Qualité aéronautique élimine les pièces les plus faibles, mais elle ne peut être confirmée par l'expérience faute d'un nombre suffisant d'essais.

En pratique, avec les écarts types sur log S_{∞} cités plus haut et les 6 essais de fatigue habituels pour les pièces importantes, on obtient les facteurs de dispersion suivants :

- Alliages légers : 1,95
- Aciers : 1,77

Ces facteurs sont utilisés pour toutes les pièces justifiées à partir d'essais de fatigue individuels. En ce qui concerne les essais complets en rotation sur réducteurs, où la surcharge aggrave les contraintes de fonctionnement, on admet les facteurs 1,4 pour 1 essai et 1,3 pour 2 essais.

2.2.2 - Fatigue oligocyclique ou temporaire :

Il s'agit, comme nous l'avons dit, de la répétition de contraintes élevées produisant l'endommagement de la pièce aux faibles nombre de cycles et que la forme des courbes de fatigue suggère de limiter aux nombres de cycles inférieurs à 10^5

Dans le domaine de l'hélicoptère qui nous intéresse, ce type de fatigue peut être produit par les charges sol-air sol telles que l'établissement puis la suppression de certaines charges moyennes (force centrifuge, couple etc...) auxquelles viennent s'ajouter le cycle vibratoire le plus élevé du vol.

Ce peut être aussi des pointes de contraintes très rares parmi les charges vibratoires pour les éléments mécaniques où l'action des manoeuvres et des rafales pour la cellule.

Pour justifier la resistance à ce type de fatigue où l'on travaille près des limites des matériaux, il ne peut être question de faire des essais en majorant les charges rencontrées en service puisque, à la limite on produirait la rupture statique de la pièce à la première mise en charge.

L'expérience montre d'ailleurs que, dans ce domaine, la sécurité prise sur le nombre de cycles est tout à fait significative et qu'à un niveau de charges donné , le logarithme du nombre de cycles suit pratiquement la loi normale

Dans la mesure où la fatigue oligocyclique intervient seule, c'est à dire lorsque les contraintes vibratoires auxquelles la pièce est soumise (aux harmoniques des différents régimes de rotation par exemple) sont inférieures à sa limite de fatigue sûre, on détermine la tenue moyenne de cette pièce en effectuant des essais de fatigue aux charges de vol.
Ces essais peuvent être conduits sous charge constante ou sous charges programmées selon la marge dont on dispose.

On passe ensuite de la tenue moyenne à tenue sûre en prenant la sécurité sur le nombre de cycles pour fixer éventuellement la limite de vie dela pièce considérée. En ce qui concerne les fuselages ou les empennages, on se conforme à l'usage des avions où le coefficient de sécurité habituel est compris entre 3 et 5. Pour les autres parties de l'appareil et en accord avec les B.C.A.R Section G, on prend les coefficients de sécurité k de la figure 10.

2.2.3 - Combinaison des deux types de fatigue :

Lorsque la pièce considérée subit à la fois des charges peu nombreuses mais proches de sa résistance statique et des charges alternées excédant sa limite de fatigue sûre, on peut employer deux méthodes de justification : la justification aux charges programmées dont nous parlerons au chapitre suivant et la justification à charges constantes.

Dans ce dernier cas, on peut déterminer la courbe de fatigue moyenne en incluant dans l'essai, au même titre que les charges constantes, les charges répétées à fort niveau (fig.11).

Pour ajuster au mieux le nombre de cycles de fatigue temporaire S_T, on opère de la manière suivante :

- On estime la durée de vie de la pièce H en heures, puis connaissant le nombre de cycles n de charges S_T par heure de vol et le coefficient de sécurité k à assurer suivant para.2.2.2, on calcule le nombre total de cycles à réaliser en moyenne pendant les essais:

$$N_T = k\ n\ H$$

- On choisit un niveau d'essai vibratoire S_V pour rompre la pièce en moyenne à N cycles et l'on répartit des blocs de fatigue temporaire S_T pour obtenir N_T cycles en fin d'essai.

C'est la méthode que nous employons généralement. Elle possède l'inconvénient de nécessiter fréquemment des compléments d'essais à charges différentes car la concommittance entre les nombres de cycles N et N_T est rarement assurée d'emblée.

On peut aussi faire les essais de fatigue temporaire seule pour en tenir compte sous forme d'endommagement horaire au moment du calcul de la durée de vie.

2.3 - Calcul de la durée de vie :

Deux méthodes sont en concurrence et possèdent chacune leurs avantages et leurs inconvénients. Elles se distinguent par le type d'essais de fatigue : charges constantes ou charges programmées.

2.3.1 - Justification à charges constantes :

Connaissant d'une part le spectre de vol et d'autre part la courbe de fatigue sûre, on calcule le dommage horaire, correspondant à chaque niveau de charge S et chaque cas de vol, égal au rapport du nombre de cycles n effectué en 1 heure de vol au nombre de cycles N correspondant à S sur la courbe de fatigue sûre.

Le dommage horaire global est obtenu en utilisant l'hypothèse de MINER qui consiste à sommer arithmétiquement les dommages élémentaires.

La durée de vie est enfin calculée en écrivant qu'elle correspond à un endommagement égal à l'unité.

Lorsque les essais de fatigue sont faits en incluant les charges temporaires, on calcule séparément les durées de vie obtenues en fatigue vibratoire et en fatigue oligocyclique et l'on retient la plus faible des deux.

Cette méthode a l'avantage de permettre le calcul des durées de vie de toutes les utilisations sans répétition des essais de fatigue.
Elle a par contre l'inconvénient de n'être juste qu'à la limite (cas de la charge en service constante) et de ne pas tenir compte des phénomènes de retard ou d'accélération des dommages dus à l'ordre d'application des cycles relativement à leur amplitude.
Elle nous donne cependant de bons résultats depuis des années et les constructeurs d'avions ont montré qu'elle est généralement un peu pessimiste.

2.3.2 - Justification à charges programmées :

Elle consiste à faire les essais en respectant l'ordre dans lequel les charges se produisent en vol tout en les majorant en amplitude ou en nombre selon qu'il s'agit d'efforts vibratoires ou temporaires. Les facteurs d'amplification, dépendant du nombre d'essais, sont ceux du paragraphe 2.2

La durée de vie est alors tout simplement la moyenne des durées de vie démontrées par chaque éprouvette.

Cette méthode permet de s'affranchir de l'hypothèse de MINER, ce qui est un avantage appréciable. Elle a cependant l'inconvénient d'être assez longue si l'on veut garder suffisamment de charges faibles pour intégrer les phénomènes de retard au dommage et surtout elle exige une série d'essais par type de mission. C'est la raison pour laquelle nous l'employons rarement.

3 - AMELIORATION DES METHODES :

Devant les difficultés rencontrées dans les essais où l'on mélange les charges vibratoires et temporaires, nous avons commencé à déterminer la tenue des pièces réelles à la fatigue oligocyclique. Ceci permet dans un premier temps de calculer séparément les dommages dus à ces deux types de charges puis à les cumuler pour faire le calcul de la durée de vie.

Il serait bien plus satisfaisant pour l'esprit de définir une courbe de fatigue moyenne entre quelques cycles et l'infini ainsi que la courbe de fatigue sûre correspondante que l'on pourrait utiliser, soit pour calculer la durée de vie à l'aide des dommages cumulatif, soit pour définir, dans le cas des essais sous charges programmées à un risque donné, le facteur d'amplification à appliquer à chaque niveau de charge.

Nous nous proposons de réaliser un programme d'essais sur éprouvettes pour définir la forme de la courbe S/N et pour chiffrer la dispersion aux faibles nombre de cycles. Le simple fait que l'adaptation plastique est plus importante aux fortes charges qu'aux charges faibles (fig. 12) indique déjà qu'il faudra effectuer plusieurs séries d'essais en faisant varier le coefficient de surcontrainte.

Il sera nécessaire, dans une 2ème étape, de vérifier que l'hypothèse de MINER est encore justifiée dans ce domaine en comparant statistiquement les valeurs obtenues par calculs basés sur essais à charge constante et les résultats des essais sous charge programmées représentatives de l'hélicoptère.

Certaines études ont déjà été faites pour les avions et il serait intéressant de voir si elles peuvent être exploitées dans notre optique. De toute façon, la somme de travail restante et les délais nécessaires pour l'exécuter sont tels qu'une coopération entre constructeurs et laboratoires est souhaitable.

4 - CAS DES STRATIFIES :

Depuis longtemps nos machines ont comporté des structures peu travaillantes telles que portes capots ou carénages en matériaux composites, mais il a fallu attendre le développement des résines époxydes pour permettre la réalisation de pièces primaires en stratifiés.

Les stratifiés modernes possèdent une excellente résistance spécifique, permettent de faire varier très facilement la distribution de masses et de raideurs des pièces pour optimiser leur comportement dynamique et ont la propriété de se dégrader progressivement et de manière visible.
Ceci a permis, il y a une dizaine d'années, de réaliser des pales de Gazelle en fibres de verre et résine qui totalisent 1 500 000 heures de vol sans problème majeur.
Comparées aux pales métalliques, elles permettent d'obtenir sans problème une durée de vie illimitée pour une masse comparable et un prix compétitif.

L'analyse de la valeur nous a maintenant conduits à réaliser des moyeux à partir de tissus et de fibres unidirectionnelles de verre imprégnés de résine époxyde. Cette construction nous a procuré l'avantage d'une réduction de masse importante, d'une diminution des prix par réduction du nombre de pièces et des temps de fabrication ainsi qu'une réduction de l'entretien.

Il a fallu cependant chercher des solutions de mécanisation pour améliorer des procédés de fabrication assez artisanaux à l'origine et résoudre les principaux problèmes rencontrés : difficultés de contrôle, recherche de l'influence du vieillissement et maintien du niveau de qualité.

4.1 - Détermination des caractéristiques de fatigue :

Tant pour les pales que pour les moyeux un programme très complet de détermination des caractéristiques physiques des matériaux a été mis en place (fig. 13)

Dans un 1er temps, on a qualifié les différents types de matériaux puis on a défini à l'aide des essais réalisés les valeurs minimales des paramètres retenus pour le contrôle de réception.

Ces travaux ont déjà été exposés par ailleurs mais, en ce qui concerne la fatigue qui nous occupe ici, on peut tirer les conclusions suivantes valables pour les composites verre-résine :

- l'endommagement en fatigue a une forte influence sur la résistance et le module d'élasticité, ce qui permet de fixer un critère de détérioration aux essais (variation de raideur),

- la forme générale des courbes S/N donnée pour les matériaux métalliques est applicable aux composites (fig. 14) et ceci même aux faibles nombres de cycles.

- la dispersion trouvée est faible (de l'ordre de grandeur de celle des pièces en acier) mais la moyenne peut présenter des variations importantes dues aux très nombreux facteurs intervenant dans l'élaboration des matériaux de base, des demi-produits et des pièces : un contrôle de qualité à la fatigue par lot est indispensable.

- L'absorption d'humidité peut réduire de 50 % la tenue à la fatigue des produits minces. Elle n'a qu'une faible influence sur les pièces massives telles que pales et moyeux, car elle n'intéresse que les couches superficielles.

4.2 - Calcul de la durée de vie :

L'expérience montre, qu'avec les pièces en stratifié, il est difficile d'obtenir des ruptures de fatigue franches. En général, les caractéristiques statiques se dégradent lentement au delà d'un certain nombre de cycles auquel apparaissent des délaminations ou des dégradations locales des fibres extérieures. On peut donc dire intuitivement que ces pièces possèdent un certain degré de "fail safe".

Cependant, faute de l'avoir démontré mathématiquement, nous avons jusqu'alors calculé les durées de vie avec les méthodes classiques exposées plus haut et dont la validité a été prouvée par essais sur éprouvettes. C'est le cas de toutes nos pales plastiques ainsi que celui des parties centrales des moyeux d'Ecureuil et de Dauphin.
Il y a certes un léger manque à gagner sur la masse de ces pièces mais la marge inhérente au procédé est appréciée pendant les premières années de mise en service de nouveaux matériels.

Les bras extérieurs de nos moyeux en matériau stratifiés ont été optimisés pour assurer le meilleur compromis entre un encombrement réduit pour diminuer la trainée, une raideur suffisante pour assurer la fonction de butée basse de pale ainsi qu'une tenue à la fatigue acceptable aussi bien en moment qu'en effort tranchant sous l'action des mouvements forcés de battement nécessaires au vol.

Ceci nous a conduits, avec les méthodes actuelles, à donner à ces pièces une durée de vie réduite quoique acceptable et qu'il était intéressant de revaloriser.

Dans ce but, la conception des bras a été menée de telle sorte que les premières détériorations se produisent en cisaillement dansle plan moyen plutôt qu'en rupture des fibres, ce qui permet un temps de propagation assez long sans dégradation rhédibitoire des caractéristiques de la pièce.

Le dimensionnement étant acquis, nous avons procédé aux essais de fatigue qui se sont déroulés en deux temps :

- dans une première phase qui est celle de détermination classique de la durée de vie, on fatigue la pièce à charge constante jusqu'à apparition des premières dégradations détectables ;

- l'essai est alors repris aux charges de vol, appliquées sous forme de programme, et l'on détermine le temps qui s'écoule entre la première dégradation détectable et la rupture de la pièce.

Nous avons ensuite déterminé, à l'aide des méthodes conventionnelles exposées plus haut, le risque R1 de début de dégradation de la pièce en fonction de la durée de vie (voir fig.15A)

Ceci s'obtient en calculant cette dernière à partir des courbes iso probabilité de rupture à différents risques.

Enfin nous avons exploité statistiquement les essais de propagation des détériorations. En supposant fixée une périodicité d'inspection, on calcule le risque R2 pour qu'un début de dégradation, survenant à un temps de vol quelconque entre deux inspections, se propage de manière catastrophique en vol avant l'inspection suivante. On trace ainsi le risque R2 en fonction de la périodicité (fig. 15B).

A noter que la 1ère série d'essais réalisée montre qu'une distribution logarithmo-normale du temps de propagation semble convenir.

En se basant sur le fait que le début de dégradation et la propagation de la rupture sont deux phénomènes indépendants, on peut dire que la probabilité totale de rupture est égale au produit R1x R2 et l'on peut choisir toute combinaison de la durée de vie et de la périodicité d'inspection qui limite le risque à la valeur acceptée (fig. 15C).

5 - CONCLUSION :

L'étude plus systématique de la fatigue oligocyclique ainsi que des recherches sur la propagation des détériorations dans les matériaux composites sont susceptibles d'apporter des augmentations importantes des durées de vie sans dégrader la sécurité.

Des programmes ont été lancés coté français mais, étant donnée l'ampleur des travaux à effectuer, une coopération entre constructeurs permettrait de gagner du temps et de réduire les coûts.

FATIGUE OF HELICOPTERS - SERVICE LIFE EVALUATION METHOD

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1. INTRODUCTION

The methods of fatigue substantiation for helicopter components have hardly progressed these last ten years except for the few refinements resulting from the evolution of measuring techniques, the accumulation of tests and the service experience acquired.

The general principle consists in first evaluating the fatigue strength of the component and determining the value and frequency of the loads to which it will be subjected during normal operation, then deriving from these data the steps to be taken to make the possible occurrence of serious accidents due to the failure of the component extremely remote :

- no action (extreme resistance)
- withdrawal from service after a certain number of hours (limited or not)
- periodicity of checks for detection of damage
- combination of the two previous limitations

All currently used methods are based upon the fact that fatigue strength depends on contingencies, its distribution being almost gaussian provided an appropriate statistical variable is selected.

The purpose is also, with a few exceptions, to protect those aircraft which operation is most straining considering the wide range of missions to be performed by a same helicopter.

Currently, the method is different for the parts mainly dimensioned by high cycle fatigue (rotors and gearboxes) and for those subjected to low cycle fatigue (e.g. fuselage).

Safety is determined from the stress in the 1st case and the number of cycles in the 2nd case.

Where both types of fatigue are encountered, various methods allow taking into account their superimposition, but each of them leads to practical problems.

These two modes of fatigue being two aspects of a same phenomenon, it seems realistic to use a single approach and select the acceptable total risk, whether the damage is calculated for each load level or the tests are conducted under programmed loads. In both cases, it would be necessary to have the equal probability-of-failure curves between a few cycles and the infinite, which has not been achieved to date.

It has been proved by now that composite materials could be given the same treatment as metals as regards fatigue strength and that their fail safety could be taken into account when the first deteriorations appear on external surfaces. Structures made from laminates have comparatively high performance as to weight or service life but they require a very strict quality audit to prevent high variations of the fatigue properties due the complexity of manufacture of these materials.

2. CURRENT METHODS FOR CONVENTIONAL PARTS

The service life of a part depends on the loads to which it is subjected during operation and on its fatigue strength. In the course of years we have tried to keep to reality in both these fields to achieve the most efficient solution - lower weight and higher durability - while retaining a satisfactory level of safety.

2.1. Working loads :

It is obviously in this field that our efforts were centred.

Estimations of flight spectra by committees consisting of the manufacturers, operators and authorities, no longer considered satisfactory, have been replaced by actual measurements during operation.

The first attempt of this kind, initiated by the French authorities for the PUMA, though seemingly costly, proved extremely cost efficient as it enabled doubling the service life of the blades and almost trebling that of essential control components such as the swashplate.

In this respect, fig. 1 shows the difference in the distribution of estimated and measured load factors, the latter being better distributed than expected in relation to speed and most of them corresponding to turns with less than 30° bank.

Also, the mechanization of processing and analysis techniques enabled replacing the previously used maximum stress of each configuration by a load histogram based upon the analysis of the signal recorded over the entire duration of the flight condition considered.

As regards manoeuvers with low repetitivity, this operation should be carried out as many times as necessary to correctly illustrate an average flight.

There again, service lives are much extended as shown on figure 2 by the dynamic load distribution on a blade during a flare out, representing the mean values of 100 manoeuvres of this type.

It is obvious that any new calculation method involving the aircraft operating strains should be implemented by experience. The question should therefore be considered as to whether the above "refinements" do not reduce dangerously the effective safety margin inherent in the former methods.

In fact, the experience (flying hours) acquired these last 10 years with this method is greater than that we previously had and no case of accident due to an insufficient fatigue margin has been recorded. Moreover, a better knowledge of the flight loads has led, in some cases, to a reduction in the service life of the parts, which is beneficial as regards safety.

It should finally be noted that service lives are greatly affected by the mission. Figure 3 gives the service life vs mission for a medium weight helicopter. Service lives are indicated by the shaded blocks, the service life in transport mission being taken as the unit.

It seems therefore recommended, from a financial point of view, to define service lives varying with the type of mission.

We only apply this to the various versions of a same helicopter and only provided the relevant parts are not mixed during overhauls ; this is the case of civil and military versions.

It has also been admitted to define an operation factor to be applied to the effective flying hours for a few major parts and in the cases where certain helicopters are intended for a specific mission (e.g. crop spraying or ASW). The study of figure 3 also shows that the service life of most components is dependent on the aircraft weight. An extensive statistical survey of the load distribution during operation would therefore be a most rewarding procedure.

2.2. Fatigue strength and safe curve

It is now admitted that fatigue strength depends on contingencies and can be represented, for a given part, by S.N.P. curves (stress or load, number of cycles, probability). If the corresponding set of curves is known (fig. 4) and if we consider the simple case of a part subjected to a constant load level, these will be no difficulties in determining the number of cycles (or flying hours) requiring withdrawal of the part in order no to exceed a given risk of failure at end of flights.

The equal probability curve corresponding to this risk is the safe curve or working curve.

The method used varies according as the considered part is mainly subjected to vibratory fatigue (low loads end large number of cycles) or to low cycle fatigue (high loads and small number of cycles).

2.2.1. Vibratory fatigue

2.2.1.1. Fatigue curve equation

The numerous fatigue test results available show that, for the numbers of cycles in excess of 10^5 - range to which we restrict the vibratory fatigue-, the equal probability of failure S/N curves can be represented by a unique typical equation dependent only on the material and on the presence of fretting corrosion.

According as the material has or not a fatigue limit, the equation is :

$$S/S_{\infty\ p} = 1 + \frac{A}{N}\gamma$$

or :

$$S/(S_N = 1)_P = \frac{1}{N}\gamma$$

where $S_{\infty\ p}$ and $(S_N = 1)_P$ represent the fatigue limit and the resistance for 1 cycle (or 1 megacycle). having a probability 1-P to be reached or exceeded for the considered family of parts.

In other words, P is the probability for a part to have a fatigue strength lower than $S_{\infty p}$ or $(S_N = 1)_p$ as appropriate.

It should be noted that the assumption of equal probability curves related to one another formulated hereabove can only be verified at high values of probability (1/10 or at least 1/100) when the number of test results available is not sufficient.

The table of fig. 5 gives the coefficients of S/N curves used for the most usual materials. They are determined through testing of a minimum of 100 coupons the shape of which is adapted to the material and the processing and analysis are effected by linear regression on logarithmic coordinates.

2.2.1.2. Selection of the statistical variable

The S/N curve having the profile as defined in the former paragraph, it is appropriate to determine the distribution law for fatigue limit S_∞ (or fatigue failure load for a given number of cycles S_N) to allow extrapolating the experimental values down to a probability sufficiently low to be accepted for operation.

For this purpose, it is necessary to select a statistical variable, a function of S or N, that is gaussian in the field considered.

Let us first note that, by the very nature of the fatigue phenomenon, the distribution is restricted to the positive values of the two coordinates and that S and N are not appropriate. Log. S and log. N yield satisfactory results for high values of S (fig. 6) but log. N is inappropriate as concerns the low values of stress S (infinite mean and standard deviation).

As we are especially concerned with low stresses in the case of the vibratory fatigue, log. S_∞ shall be selected as statistical variable ; fig. 7, drawn from the stastical quality control fatigue testing of Alouette III blades, shows that this assumption has been implemented by experience.

2.2.1.3. Working fatigue curve

The curve must be determined from fatigue tests and requires that the mean fatigue limit and the standard deviation for the part considered be known.

A research committee supervised by the french STAé worked on the analysis of numerous tests carried out on the Alouette blades and reached the following conclusions :

- the mean for a family of parts is well defined from six test points (within ± 5 % approx. for a normal dispersion)
- a very large number of tests are required for a correct estimation of the standard deviation.

Consequently, the mean fatigue limit can be determined from tests on actual parts whereas the standard deviation should be assessed from the former experience acquired with a large family of similar parts.

As regards the blades, the standard deviation has been assessed to approximately 0,06 for light alloys and 0,045 for steels, and these values have been extended to all the aircraft parts.

Once the sample mean and standard deviation are known, an acceptable value of risk should be selected to allow inferring the dispersion factor defining the relation between the mean and safe curves.

The calculation is based upon the SOULEZ-LARIVIERE method which assumes a failure risk of 10^{-6} with a truncated law of probability to reduce to zero the probability of failure at approximately one third of the fatigue limit.

The number of standard deviations of the sample to be entered under the mean curve is calculated from the number of tests available using HALD'S formula which takes into account the mean and standard deviation errors with a degree of confidence of 90 %, as shown on fig. 8.

The working fatigue limit thus obtained, expressed as % of mean limit, is then corrected as shown on fig. 9.

This truncation of the law of probability corresponds to the common usage by which no experimental substantiation is required when the maximum flight load is less than one third of the fatigue limit. It corresponds to the principle by which the aeronautical Quality Control discards the weakest parts but cannot be implemented by experience for lack of a sufficient number of tests.

Practically, using the standard deviations on log. S_∞ as quoted above and the 6 fatigue tests usually performed on major parts, the following dispersion factors can be obtained :

- Light alloys : 1,95
- Steels : 1,77

These factors are used for all the parts substantiated from individual fatigue tests. As regards the complete tests performed in rotation on gear boxes and in which the overload increases the working stresses, the factors 1,4 and 1,3 are admitted for one test and two tests respectively.

2.2.2. Low cycle fatigue

As said before, this fatigue consists in the repetition of high stresses entailing the deterioration of the part at low numbers of cycles ; the shape of fatigue curves suggest that this repetition be restricted to numbers of cycles lower than 10^5.

As far as helicopters are concerned, this type of fatigue can be produced by ground-air ground loads, as the successive application and suppression of certain constant loads (centrifugal force, torque etc...), to which is added the highest flight vibratory cycle.

It can also be produced by unusual stress peaks among the vibratory loads for mechanical components or by the effect of manoeuvres or gusts for the airframe.

To substantiate the resistance to this type of fatigue where materials are subjected to loads close to their strength limits, there would be no point in performing the tests while increasing the normal working loads as this would only result in the static failure of the part at the first application of the load.

Besides, experience in this field shows that safety gained on the number of cycles is extremely significant and that, at a given load level, the logarithm of the number of cycles follows almost the gaussian law.

Where only the low cycle fatigue is involved, i.e. when the vibratory stresses to which the part is subjected (e.g. at the harmonics of the various rotation speeds) are lower than its working fatigue limit, the mean strength of the part is determined from fatigue tests performed under normal flight loads. These tests can be carried out using the constant load or programmed load methods depending on the margin available.

The working strength is then derived from the mean strength by applying the safety coefficient to the number of cycles in order to determine the service life of the part considered.As regards fuselage or empennage sections, the safety coefficient normally used on airplanes applies ; this coefficient varies from 3 to 5. As concerns the other helicopter parts, and in accordance with B.C.A.R. section G, the safety coefficients K are given by figure 10.

2.2.3. Combination of the two types of fatigue

When the part considered is subjected at the same time to loads in small numbers but close to its static strength limit and to alternating loads exceeding its working fatigue limit, two substantiation methods can be used ; the programmed load method, dealt with in the subsequent chapter, and the constant load method.

In the latter case, the mean fatigue curve can be determined by including in the test the hight level repeated loads on the same grounds as the constant loads (fig. 11).

In order to best adjust the number of cycles of low cycle fatigue S_T, it is recommended to proceed as follows :

- Estimate the service life H of the part in hours then, knowing the number of cycles n of loads S_T per flying hour and the safety coefficient k to be applied as per para. 2.2.2., calculate the average total number of cycles to be carried out during the tests :

$$N_T = k\ n\ H$$

- Select the appropriate vibratory test level S_V to break the part at an average of N cycles and distribute the temporary fatigue loads S_T to obtain N_T cycles at end of test.

This is the method that we mostly use. Its main drawback is to frequently require additional tests under different loads as the numbers of cycles N and N_T seldom coincide at the first attempt.

It is also possible to carry out the sole temporary fatigue tests and use the results in the calculation of the service life after conversion into damage per hour.

2.3. Calculation of the service life

Two methods are available, both with advantages and drawbacks. They only differ by the type of fatigue tests : constant loads or programmed loads.

2.3.1. Substantiation by the constant load method

The flight spectrum and the working fatigue curve being known, the method consists in calculating the damage per hour, corresponding to each load level S and to each flight condition, as the ratio of the number of cycles n performed in 1 flying hour to the number of cycles N corresponding to S on the working fatigue curve.

The total damage per hour is obtained through an arithmetical sum of the individual damages according to MINER'S assumption.

The service life is finally calculated by writing that the total damage reaches unity.

When the fatigue tests include the low cycle loads, the service life is calculated separately for the vibratory fatigue and the low cycle fatigue, and the lower value is retained.

This method is particularly suitable as it allows calculating the service life corresponding to the various missions without repeating the fatigue tests.

However, its main drawback is that it is accurate at the limit only (case of the constant working load) and does not take into account the phenomena of over stressing and under stressing due to the cycle application sequence with respect to the cycle amplitude.

This method has been successfully used for many years and the aircraft manufacturers have shown it to be slightly pessimistic.

2.3.2. Substantiation by the programmed loading method

The method consists in carrying out the tests according to the sequence in which the loads are applied during the flight while increasing their amplitude or number according as the stresses are vibratory or low cycle. The amplification factors, dependent on the number of tests, are those of para. 2.2.

The service life is then calculated as the mean of the service lives demonstrated for each test speciment.

This method is particularly suitable as it allows giving up MINER's assumption. Its main drawback however is that it is rather long when it comes to retaining sufficient low loads to integrate the phenomena of under stressing and that it requires a series of tests for each type of mission. This is the reason why we seldom use it.

3. IMPROVEMENT OF METHODS

Considering the difficulties encountered during the tests involving both vibratory and low cycle loads, we began determining the low cycle fatigue strength on effective parts. This allows first calculating separately the deteriorations due to the two types of loads, then cumulating them to calculate the service life.

It would be of utmost interest to define a mean fatigue curve between a few cycles and the infinite as well as the corresponding working fatigue curve to be used either for calculating the service life from the cumulated damages, or for defining the amplification factor for each load level and for a given risk in the case of programmed loading.

We intend to develop a testing programme on specimens to define the profile of the S/N curve and work out the dispersion al low numbers of cycles. Since the plastic flow properties are greater under high loads than under low loads (fig. 12); it will be necessary to perform several series of tests while varying the stress concentration factor.

It will also be necessary in a second step, to check whether MINER's assumption still stands in this field by statistically comparing the values obtained by calculations from tests under constant loading with the results of tests under programmed loading typical of the helicopter.

Some studies have already been carried out for airplanes and it would be interesting to check to what extent they could be applied to helicopters. Anyway, the remaining amount of work and the time required to carry it out are such that a cooperation of manufacturers and laboratories is advisable.

4. LAMINATES

Our helicopters have long included low stressed structures such as doors, cowlings or fairings made from composite materials, but only the development of epoxy resins enabled manufacturing primary parts from laminates.

Advanced laminates have an excellent specific strength ; they allow varying extremely easily the distribution of weights and stiffnesses in the parts to optimize their dynamic behaviour ; moreover they have the peculiar quality of deteriorating progressively and perceptibly.

These properties led to the manufacture, some ten years ago, of Gazelle blades from fibreglass and resin ; these blades have totalled up to 1 500 000 flying hours without meeting with any major trouble.

Compared to metal blades, they allow achieving without problem an unlimited service life with an equivalent weight and a better price.

The value analysis has now led us to manufacture hubs from glass fabrics and unidirectional glass fibres impregnated with epoxy resin. This design allowed reducing the weight significantly, reducing the cost through a reduction of the number of parts and of the production times, as well as reducing the maintenance effort.

It has however been necessary to seek out mechanized solutions to improve the rather elementary manufacturing processes currently used and solve the major problems encountered : inspection problems, determination of the effect of ageing and maintenance of the quality level.

4.1. Determination of fatigue characteristics

An extensive programme for the determination of the physical properties of materials has been developed both for the blades and the hubs (fig. 13).

In a first step the various types of materials have been certified then the minimum values of the parameters selected for acceptance check have been defined from the various tests performed.

This work has already been described elsewhere but, as regards fatigue, the following conclusions applying to glass resin composites can be drawn :

- fatigue deteriorations greatly affect the material strength and modulus of elasticity, which allows defining a deterioration criterion for the tests (stiffness variation)
- the general S/N curve profile given for metallic materials applies to composites (fig. 14) even at low numbers of cycles
- the scatter recorded is narrow (similar to that of steel parts) but the mean may show important variations due to the great number of factors involved in the preparation of the basic materials, semi finished products and detail parts : <u>every batch of parts must be subjected to a fatigue strength quality control</u>.
- The absorption of moisture can reduce by up to 50 % the fatigue strength of thin products. It has little influence on solid parts such as blades and hubs as it only affects the surface layers.

4.2. Calculation of the service life

Experience has shown that it was extremely difficult to obtain plain fatigue failures with laminate parts. Usually, the static characteristics begin deteriorating slowly after a certain number of cycles at which local delaminations or degradations of the external fibres appear. It can therefore be assumed that these parts have a certain degree of "fail safety".

However, as we had not proved it mathematically, we have, up to now, calculated the service lives using the conventional methods described above and which effectivity has been demonstrated by tests performed on specimens. It has been the case for all our plastic blades and for the centre sections of the Ecureuil and Dauphin hubs.
There is most certainly something to be done to improve the weight of these parts but the margin inherent in the process is greatly appreciated during the first years of operation of new material.

The arms of our laminate hubs have been optimized to achieve the best compromise between smaller dimensions to reduce drag, sufficient stiffness to provide a droop stop for the blade, and satisfactory fatigue strength as regards both moment and shear stress under the action of forced flapping movements required during the flight.

This led us, using the present methods, to define small but acceptable values of life for these parts but showed the need for their improvement.

For this purpose, the arms have been designed so that the first deteriorations appear in shear in the middle plane rather than by failure of glass fibres, which allows extending the propagation time without redhibitory degradation of the part properties.

The problem of dimensions being settled, we proceeded to the fatigue tests in the two following steps.

- during the first phase corresponding to the conventional determination of the service life, the part is fatigue tested under a constant load until the first perceptible deterioration appears.

- in the second phase, the fatigue tests are repeated under the normal flight loads applied as a programme, and the time between the first perceptible deterioration and the failure of the part is then recorded.

Using the conventional methods described above, we then determined the risk R1 of first appearance of part deterioration with respect to the service life (see fig. 15A).

This is obtained by calculation of the service life from the equal probability of failure curves at various risk levels.

We finally performed a statistical survey of the deterioration propagation tests. Assuming a fixed inspection interval, the risk R2 is calculated for a deterioration, initiated after any number of flying hours between two inspections, to propagate in a catastrophic way in flight before the next inspection. Risk R2 is then plotted versus inspection interval (fig. 15 B).

It should be noted that the first series of tests performed shows that the logarithmo-normal distribution of the propagation time seems appropriate.

Assuming that the deterioration initiation and the propagation of failure are two independent phenomena, the total probability of failure is equal to the product of R1 by R2 and any combination of service life and inspection interval can be selected that limits the risk to the accepted value (fig. 15 C).

5. CONCLUSION

A more systematic study of the low cycle fatigue as well as a survey of the propagation of deteriorations in composite materials will probably allow extending significantly the service lives without impairing safety.

The French have started programmes in this direction but, considering the amount of work to be done, a cooperation between manufacturers would allow saving time and money.

DISTRIBUTION OF LOAD FACTORS IN LEVEL FLIGHT

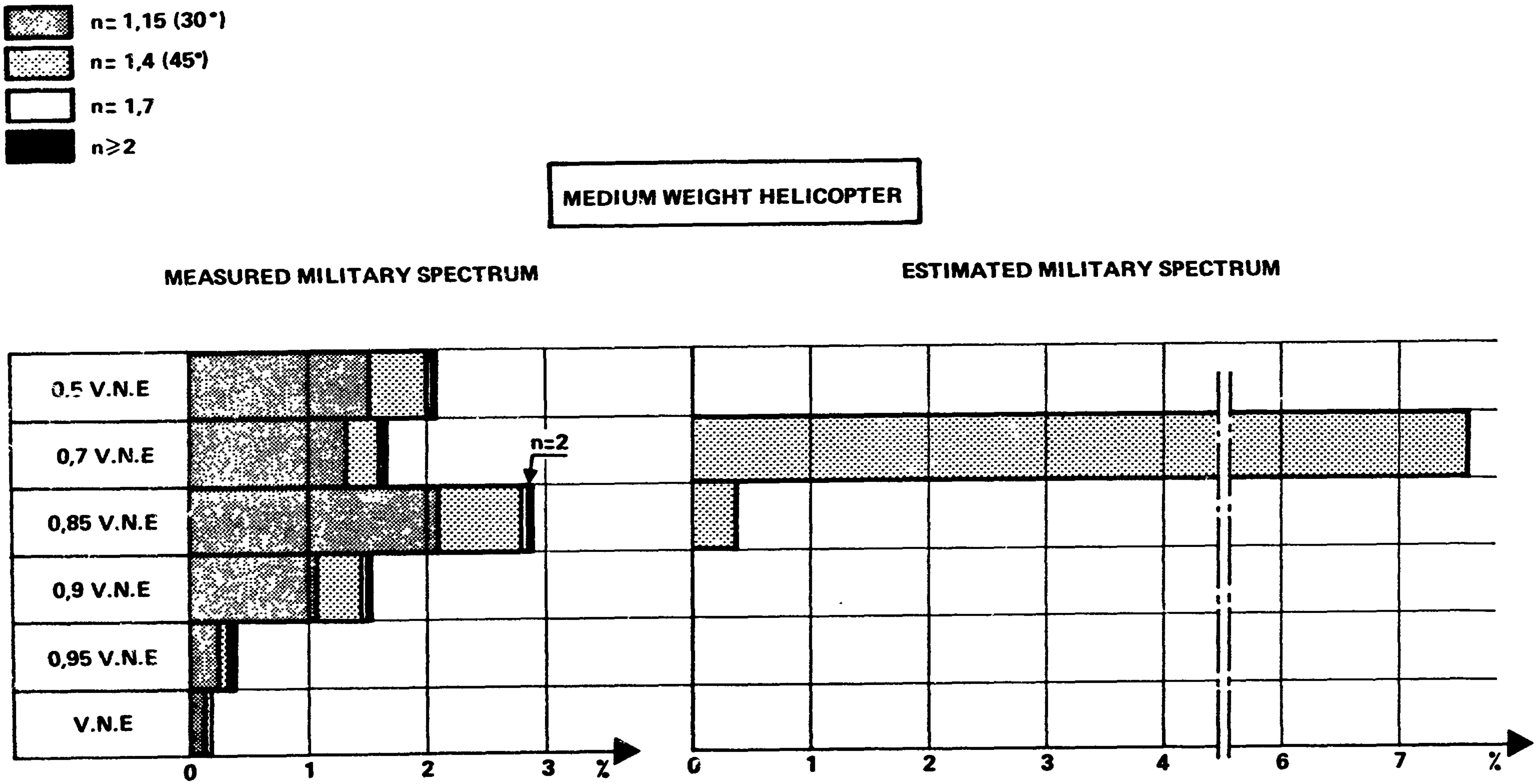

Figure 1

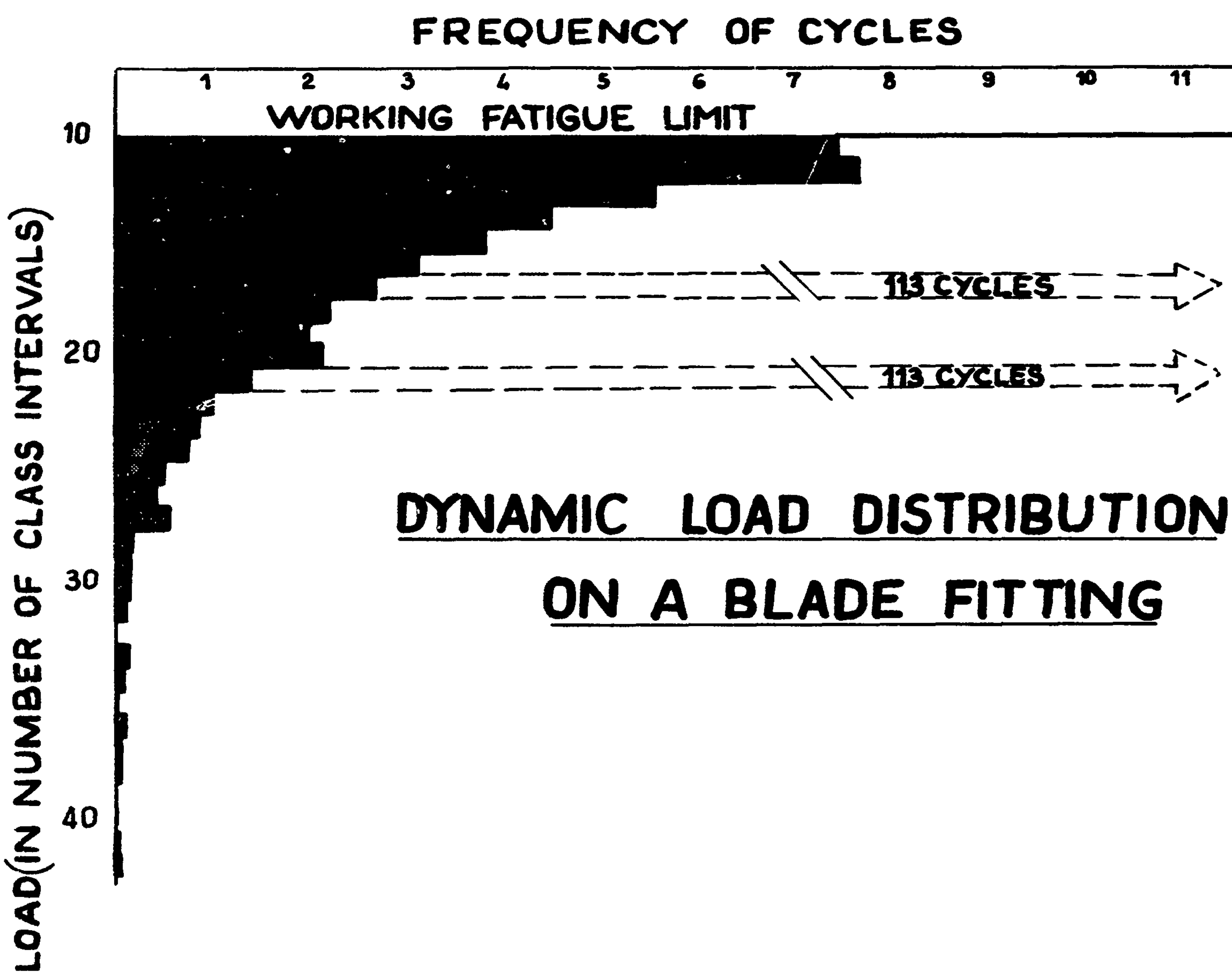
FREQUENCY OF CYCLES
1
2
3
4
5
6
7
8
9
10
11
WORKING FATIGUE LIMIT
10
20
30
40
LOAD (IN NUMBER OF CLASS INTERVALS)
113 CYCLES
113 CYCLES
DYNAMIC LOAD DISTRIBUTION
ON A BLADE FITTING

Figure 2

INFLUENCE OF SPECTRUM ON SERVICE LIFE

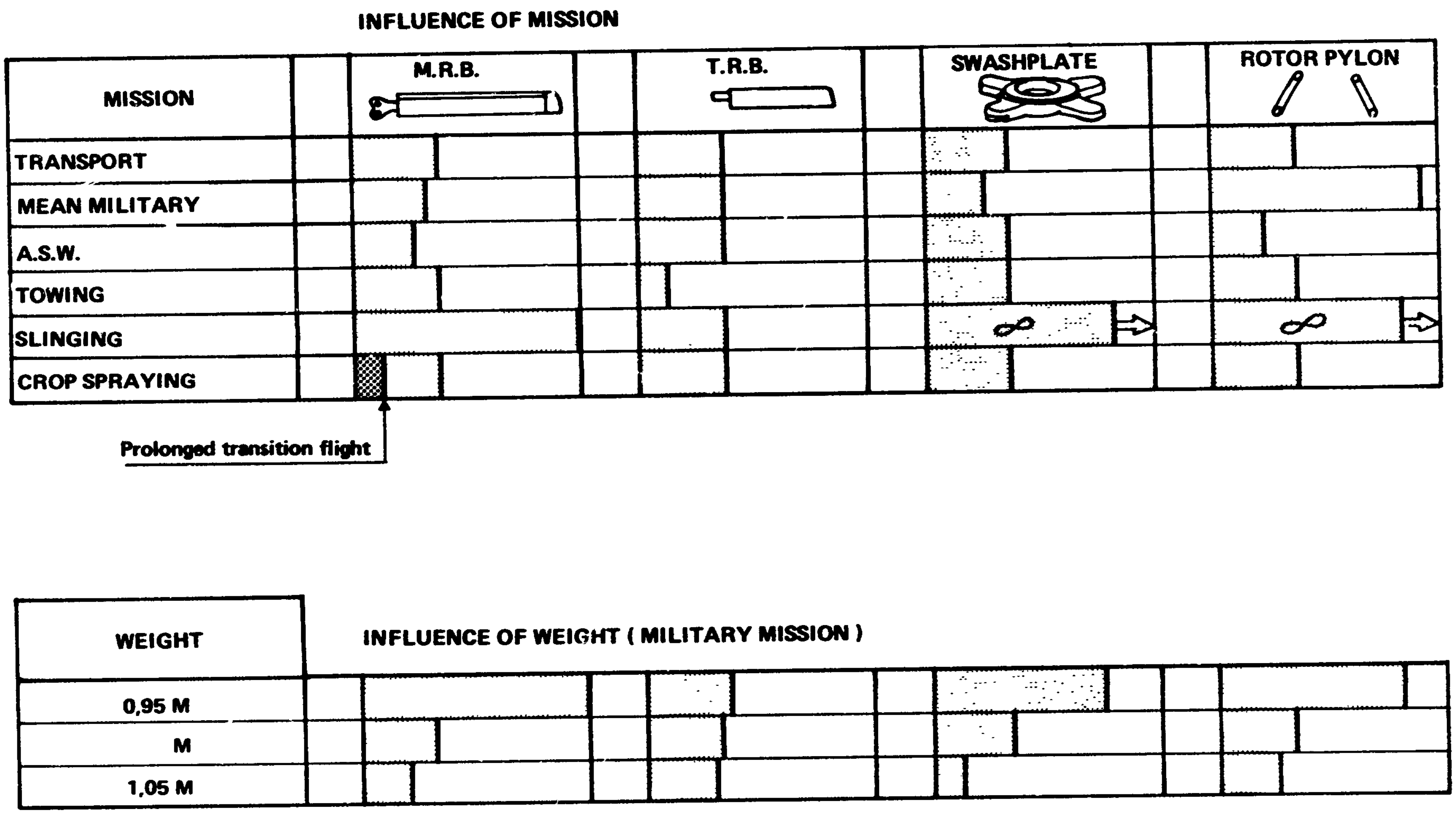

Figure 3

S.N.P. CURVES

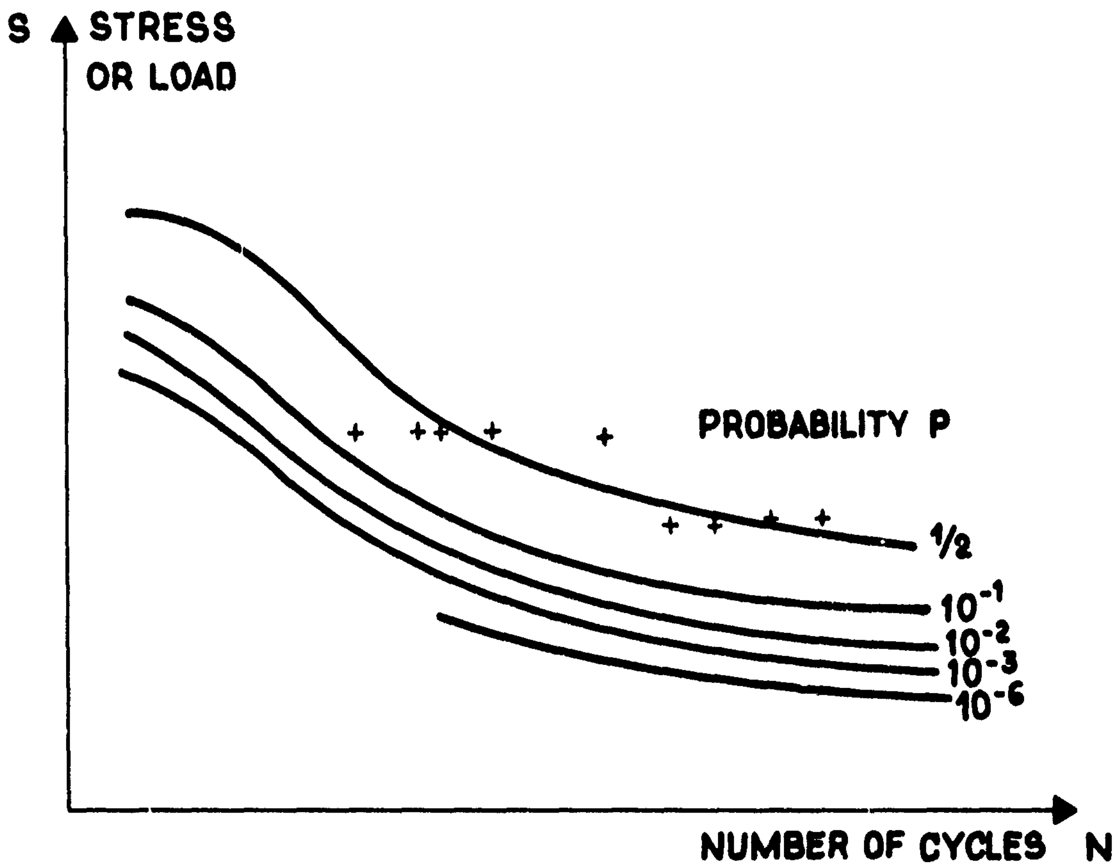

Figure 4

COEFFICIENTS OF S/N CURVES FOR VARIOUS MATERIALS

$$\frac{S}{S_\infty} = 1 + \frac{A}{N^\gamma} \quad \text{OR} \quad S = \frac{B}{N^\gamma} = \frac{S_{(N=1)}}{N^\gamma}$$

MATERIAL	PRESENCE OF FRETTING CORROSION	A *	γ
ALUMINIUM AND MAGNESIUM ALLOYS	NO	0.483	0.5
	YES	0.892	0.5
STEEL	NO	0.0323	1
	YES	0.526	0.66
TITANIUM ALLOY TA 6V	NO	0.205	0.49
	YES	0.811	0.63
GLASS ROVING	BOTH	–	0.1

* FOR N IN MILLION CYCLES

Figure 5

CHOICE OF PROBABILITY VARIABLE

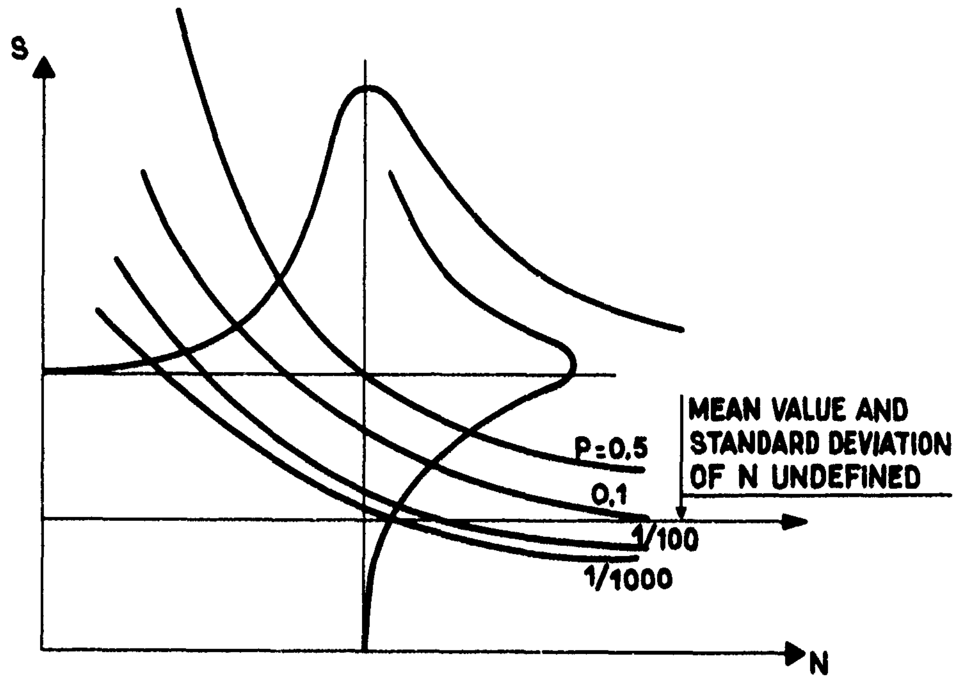

Figure 6

HENRY DIAGRAM FOR QUALITY CONTROL TESTS OF ALOUETTE III M.R. BLADE

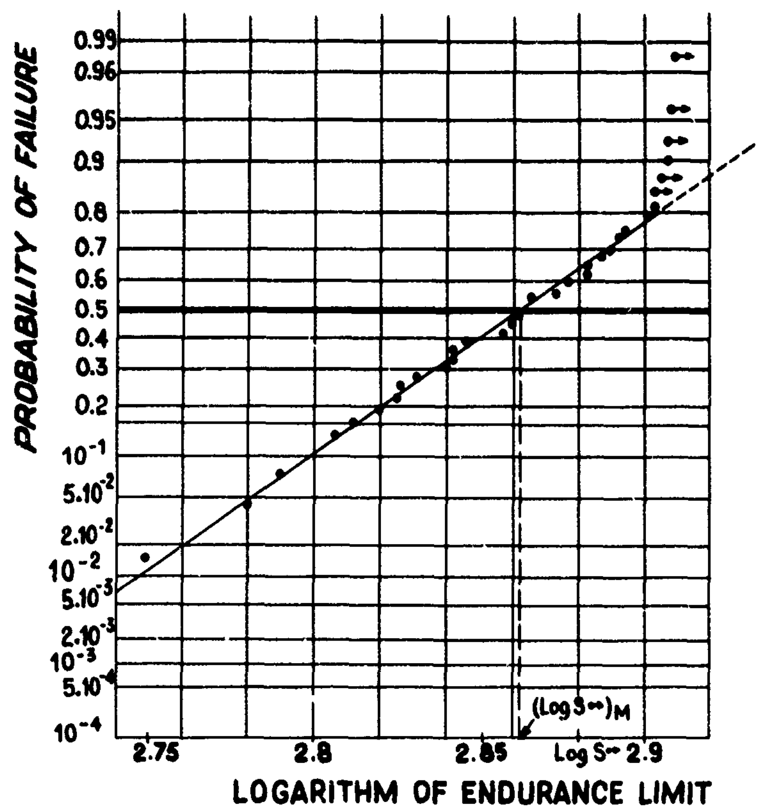

Figure 7

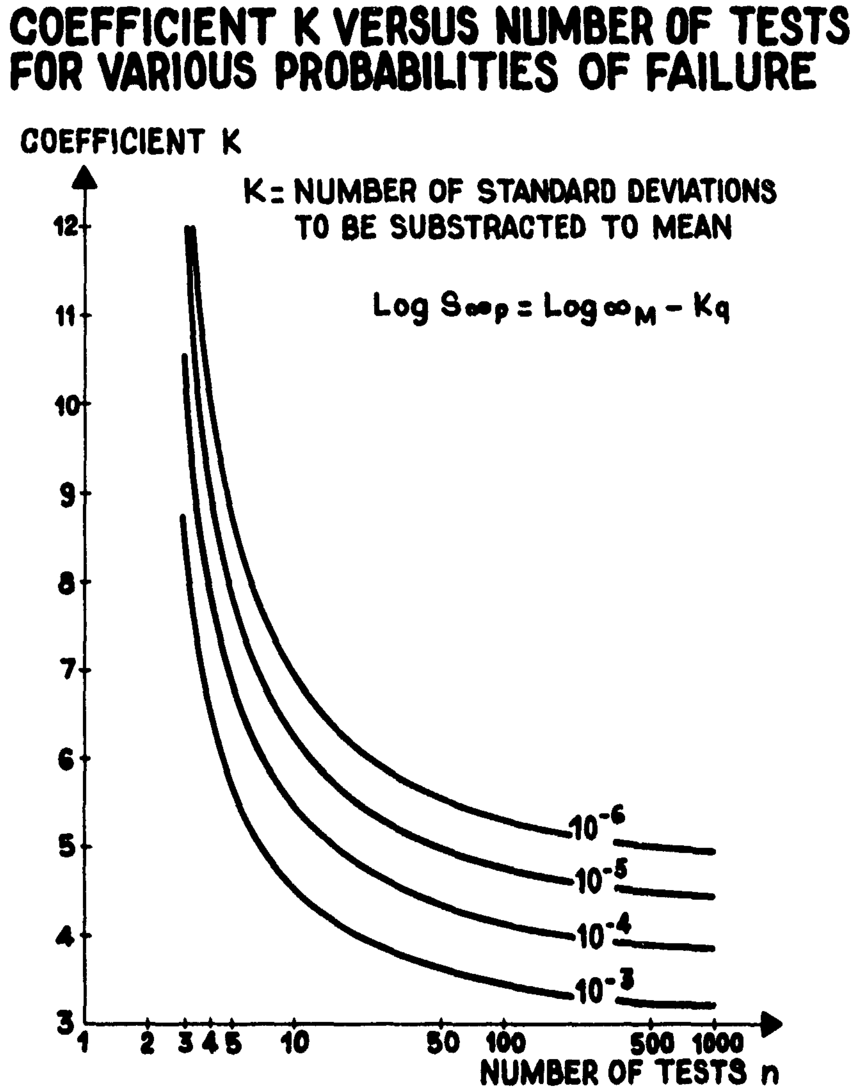

Figure 8

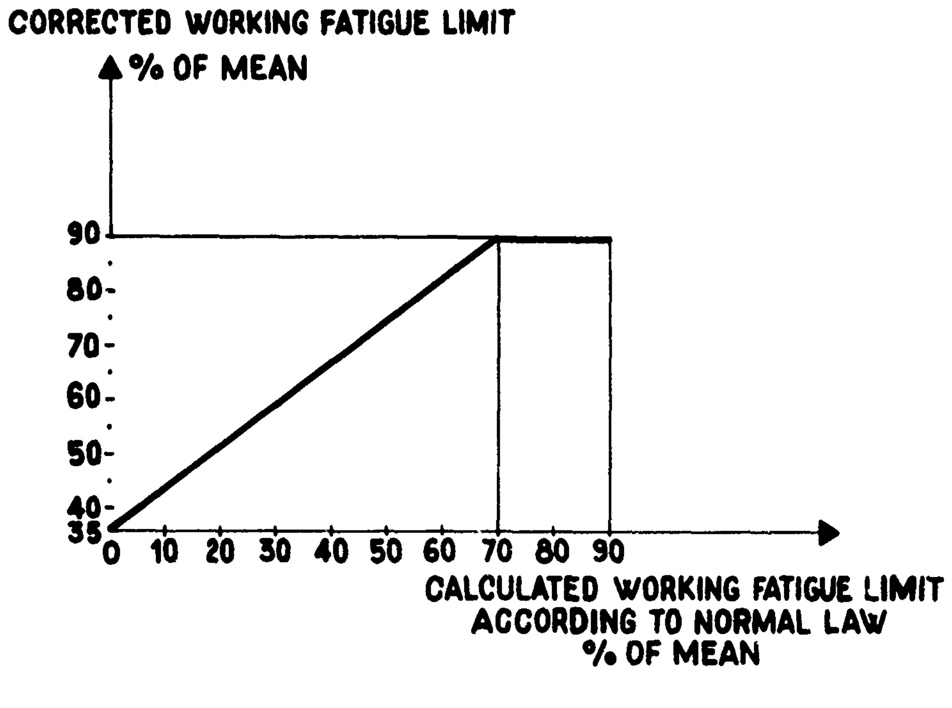

Figure 9

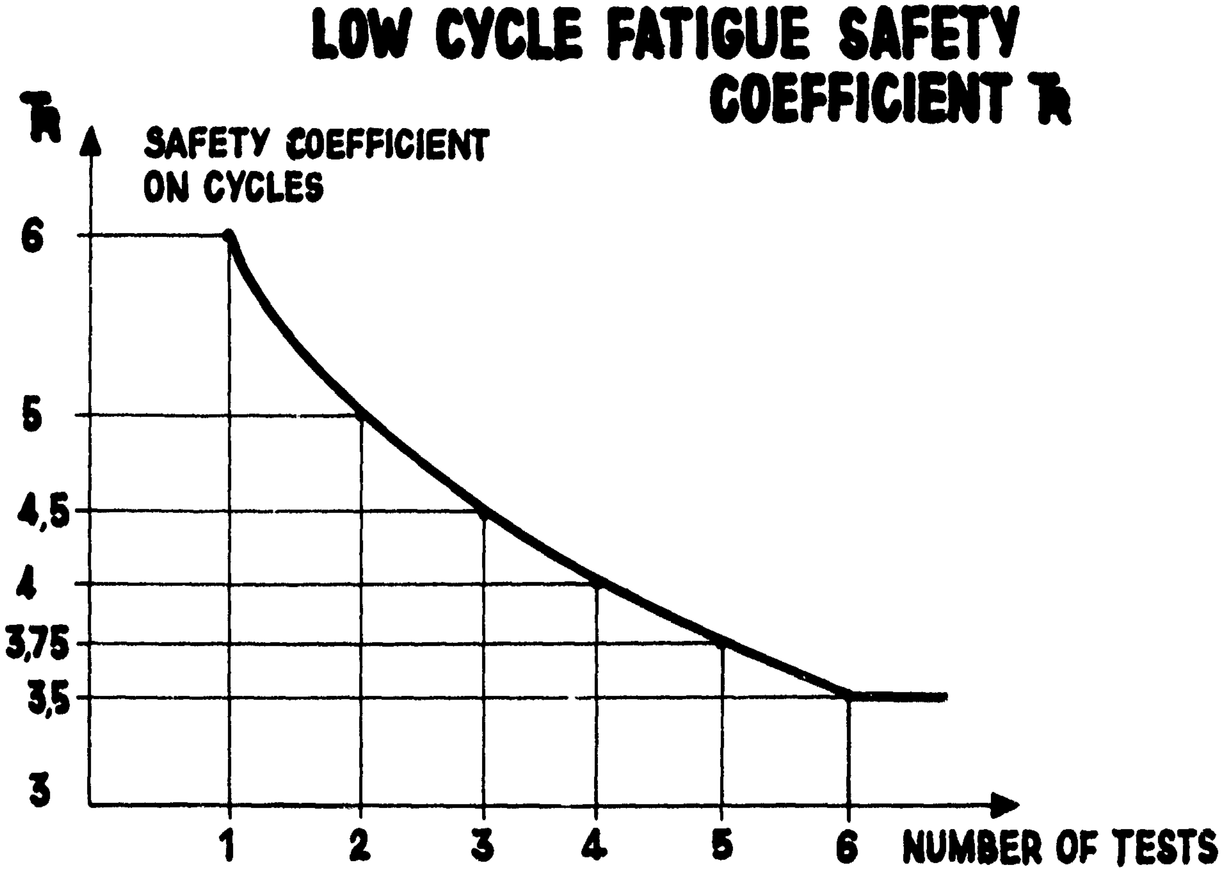

Figure 10

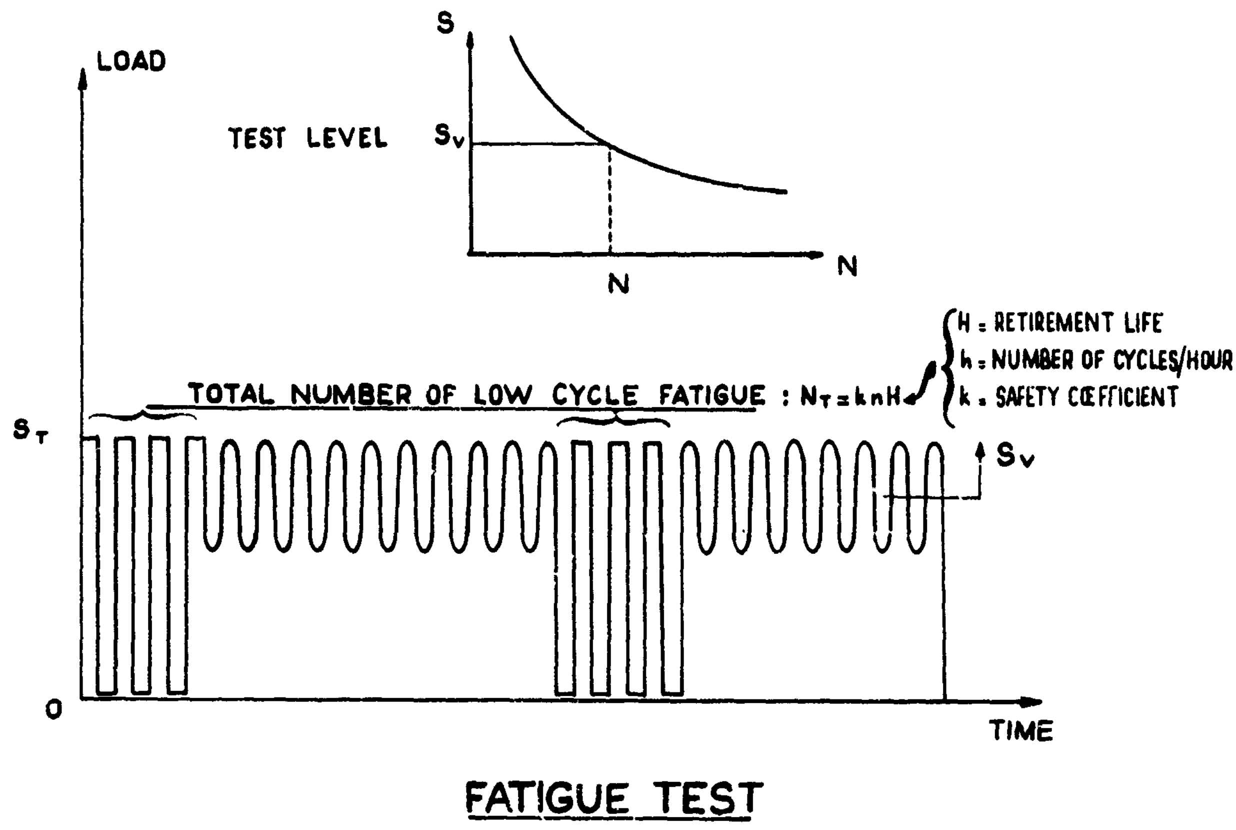

Figure 11

INFLUENCE OF STRESS CONCENTRATION FACTOR ON S/N CURVE

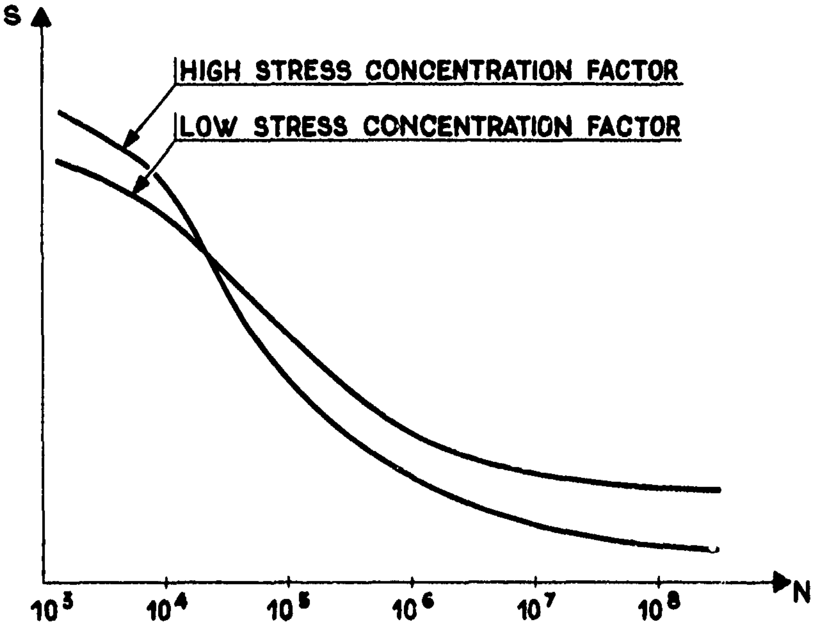

Figure 12

TESTS PERFORMED FOR COMPOSITES

ACCEPTANCE TESTS
QUALIFICATION TESTS

	TEST TEMP	WARP	WEFT
TENSION	+ 20°C	X	X
COMPRESSION	+ 20°C	X	X
BENDING	− 55°C	X	X
	+ 20°C	X	X
	+ 80°C	X	X
FLEXURAL MODULUS	+ 20°C	X	X
INTERLAMINAR SHEAR	− 55°C	X	X
	+ 20°C	X	X
	+ 80°C	X	X
AGEING STRENGTH EXPOSURE : 1000 Hrs. RELATIVE HUMIDITY : 100 %. TEMPERATURE : 70 °C	VARIATION IN WEIGHT AND VOLUME		
	BENDING AT 20 °. WARP DIRECTION		
	BENDING AT 20 °. WEFT DIRECTION		
IMMERSION IN WATER : 48 Hrs. AT 100° C	BENDING AT 20° WARP AND WEFT DIRECTIONS		
IMMERSION FOR 750 HOURS, AT AMBIENT TEMPERATURE, IN FLUIDS (TO BE DETERMINED ACCORDING TO ENVIRONMENT)			
INFRA-RED ABSORPTION SPECTRUM			
MICRO CALORIMETRY			
ON PRE – IMPREGNATED MATERIAL	VOLATILE CONTENTS		
	RESIN CONTENTS		
	UNIT WEIGHT		
ON CURED MATERIAL	COMBUSTION LOSS		
	UNIT WEIGHT		
	THICKNESS		
FATIGUE TESTS			
CREEPAGE TESTS			

Figure 13

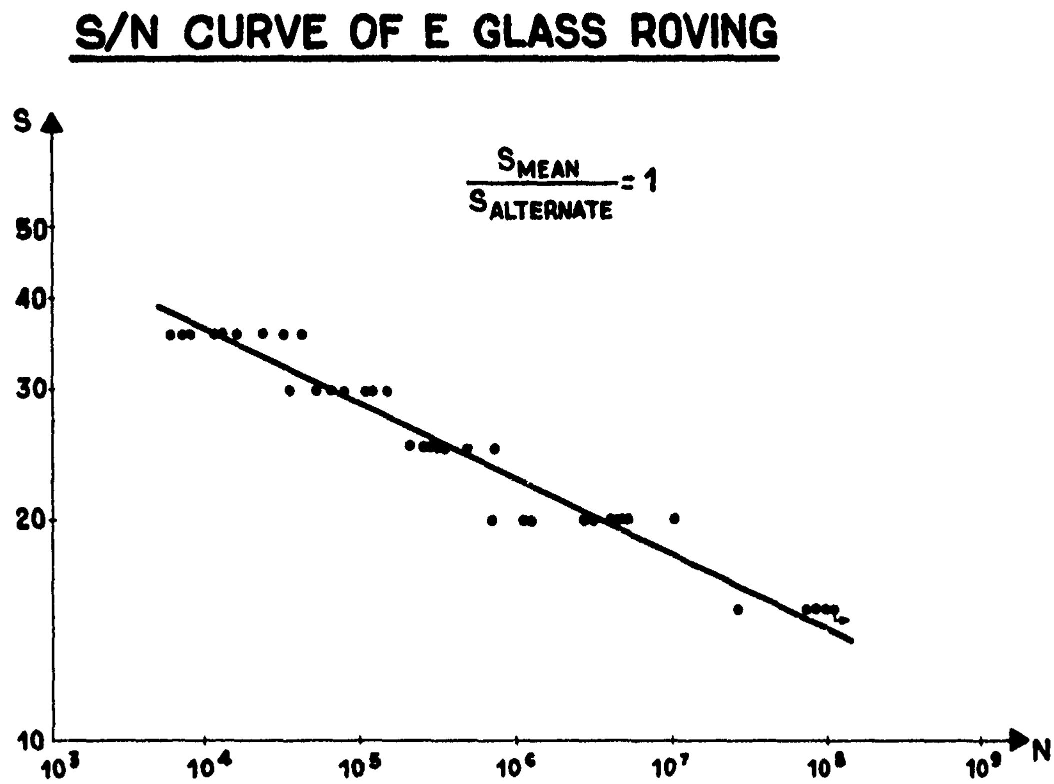

Figure 14

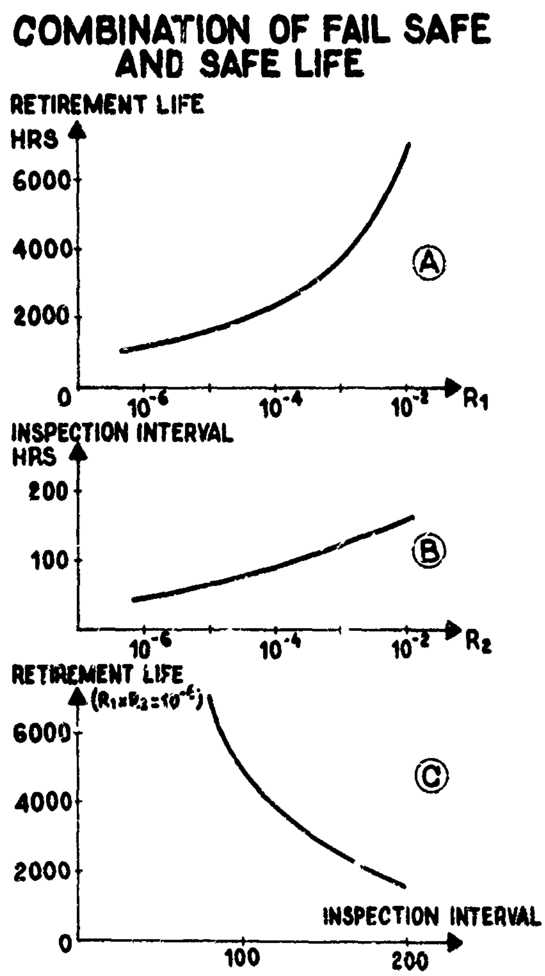

Figure 15

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Reviews	Design criteria
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14. Abstract

Helicopter designers have always been concerned with fatigue phenomena and the experience developed in this field in NATO countries has resulted in fatigue requirements which appear to have very similar objectives and philosophies.

This Report may be considered as a detailed and valuable review of current fatigue requirements and substantiation procedures in the United States, United Kingdom, Germany, Italy and France in the field of Helicopter Fatigue.

Fatigue specialists are aware of the rather uncomfortable situation which is reflected through these papers; although general requirements and specifications seem to be very similar, approved procedures applied by manufacturers may sometimes appear to be rather arbitrary or, in some cases, to differ significantly from one firm to another.

The material collected in this publication must be considered as a helpful survey to be used by helicopter specialists with a view to intensifying cooperative action within the NATO community towards improvement, rationalization and standardization of helicopter service life prediction.

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